

**NASA-AMES STANFORD UNIVERSITY
JOINT INSTITUTE FOR AERONAUTICS AND
ACOUSTICS**

AERO NO.97-52

NASA GRANT NCC 2-55

**THE RESEARCH AND TRAINING ACTIVITIES FOR THE JOINT
INSTITUTE FOR AERONAUTICS AND ACOUSTICS**

**Submitted to the
NASA Ames Research Center
Moffett Field, CA 94035**

**For a period of One Year
October 1, 1997 to September 30, 1998**

by the

**Department of Aeronautics and Astronautics
Stanford University
Stanford, California 94305**

**Principal Investigator
Professor Brian Cantwell**

September 1997

8/27/97

TABLE OF CONTENTS

ABSTRACT

1. JOINT INSTITUTE PROGRAM OVERVIEW

1.1. Introduction

1.2. Research Project Summaries

1.3. Institutional Support

1.4. Training Activities

1.5. Research Participation

2. DETAILED PROJECT DESCRIPTIONS

Project 1 - Active flow control

Project 2 - LES as a tool for studying jet aero-acoustics

Project 3 - Research on a lifting wing-flap configuration

Project 4 - Luminescent paint for aerodynamic measurement

Project 5 - Prediction of wing maximum lift for preliminary design

Project 6 - Acoustic wave scattering through a turbulent mixing layer

3. PERSONNEL

4. FUNDING

ABSTRACT

This proposal requests continued support for the program of activities to be undertaken by the Ames-Stanford Joint Institute for Aeronautics and Acoustics during the one-year period October 1, 1997 to September 30, 1998. The emphasis in this program is on training and research in experimental and computational methods with application to aerodynamics, acoustics and the important interactions between them. The program comprises activities in active flow control, Large Eddy Simulation of jet noise, flap aerodynamics and acoustics, high lift modeling studies and luminescent paint applications. During the proposed period there will be a continued emphasis on the interaction between NASA Ames, Stanford University and Industry, particularly in connection with the noise and high lift activities.

The program will be conducted within the general framework of the Memorandum of Understanding (1976) establishing the Institute, as updated in 1993. As outlined in the agreement, the purposes of the Institute include the following:

- To conduct basic and applied research.
- To promote joint endeavors between Center scientists and those in the academic community.
- To provide training to graduate students in specialized areas of aeronautics and acoustics through participation in the research programs of the Institute.
- To provide opportunities for Post-Doctoral Fellows to collaborate in research programs of the Institute.
- To disseminate information about important aeronautical topics and to enable scientists and engineers of the Center to stay abreast of new advances through symposia, seminars and publications.

The program described above is designed to address future needs of NASA Ames and has been the basis of discussion among Professors B. Cantwell, I. Kroo, S. Lele and S. Rock from the Stanford faculty and several members from NASA Ames including Dr. S. Davis, Dr. R. Mehta and Dr. S. Smith. Coordination of this activity at Ames is the responsibility of the Institute Associate Director for Center Affairs, Dr. C. A. Smith.

1. JOINT INSTITUTE PROGRAM OVERVIEW

1.1 INTRODUCTION

Experimental and computational aerodynamics have for many years played an important role in the basic and applied research programs of Ames Research Center and in the research and training activities of Stanford University. Recently, computational tools have been brought to bear on the difficult problem of flow generated noise. The coordinated use of a combination of experimental and computational tools has long been recognized as an essential part of a comprehensive approach to improving our fundamental understanding of complex flow phenomena. Developments in computational capabilities, in flow visualization, in measurement and in new kinds of wind-tunnel instrumentation will constitute a major step forward in the ability of scientists and engineers to advance the state of the art in aerodynamic design technology.

It is therefore the general character of the proposed program that it involves both experiment and computation and that these are used in complimentary ways. This approach can be undertaken only if highly qualified personnel and good research facilities are available. In this regard the blending of resources from Stanford and Ames is an important ingredient and was one of the motivating reasons behind the establishment of the Ames-Stanford Joint Institute.

In the experimental parts of the program described below, smaller scale investigations undertaken at Stanford are coordinated with both computations and experiments carried out in the more powerful facilities at Ames Research Center.

1.2 RESEARCH PROJECT SUMMARIES

The research directions summarized here, and further elaborated in the Program Description, are the result of several discussions with research management and staff at Ames Research Center. The activities are consistent with the emphasis on acoustics and high-lift in current NASA programs.

Project 1 - Active flow control

This is a continuing program in the use of active flow control as a means of regulating aircraft attitude at high angles of attack. The combined roll and yaw control of a generic thin delta wing aircraft using fore-body tangential blowing is being investigated. Techniques for developing nonlinear optimum control laws are being established using results obtained from a unique free-to-roll, free-to-yaw support system. Wind tunnel data and numerical simulations are being used to provide the aerodynamic information necessary in the formulation of control laws for this configuration.

Project 2 - LES as a tool for studying jet aero-acoustics

New subsonic and supersonic aircraft are required to meet increasingly more stringent environmental noise regulations. Current design/analysis tools for estimating the noise generated by an aircraft configuration rely strongly on

empirical formulations. With recent advances in computational technology it seems possible that important components of aircraft noise could be predicted by a theoretical approach. Since noise is generated by unsteady flow it becomes necessary to accurately predict the unsteady flow. The proposed research seeks to evaluate and develop Large Eddy Simulation (LES) as a computational technology for predicting jet-noise.

Project 3 - Research on a lifting wing-flap configuration

The adoption of lower tolerance international, national and local noise regulations and the advent of large, high lift commercial aircraft has led to a renewed interest in noise generation by airframe components. Recent studies of airframe noise have identified the wing and flap trailing edge as well as the flap side-edge as areas of elevated noise generation. In this project the fluid dynamic processes associated with these two classes of noise sources are being investigated. A NACA 63-215 Mod B airfoil section has been used by NASA Ames investigators for high Reynolds number experiments in the Ames 7x10 tunnel. These experiments included noise studies carried out by Boeing and Ames investigators using Boeing-developed phased array instrumentation. This same geometry is also being studied in CFD computations by Ames and Stanford investigators and in small scale experiments at Stanford University. The Stanford experiments emphasize: mapping the mean flow and turbulence quantities in the near wake of the flap side-edge; making unsteady pressure measurements over the flap and other sections of interest; and performing visualization and measurement of unsteady aspects of the flow which can not be easily studied in either the computations or the 7x10 experiments.

Project 4 - Luminescent paint for aerodynamic measurement

This project employs luminescent (pressure sensitive) paint to measure the spatial pressure distribution on a wind tunnel model. The emphasis is on extending the technique to low speed flows. Pressure sensitive paints are based on a class of chemicals known as porphyrins and make use of a surface reaction which, under illumination with ultraviolet light, causes the scattered light intensity to be proportional to the partial pressure of oxygen at the painted surface. The variation in scattered intensity can be recorded with a video camera and used to infer surface pressure over an extended area. With further development these paints, along with similar systems capable of measuring wall shear stress, promise to revolutionize wind tunnel testing techniques. In particular, the high cost of pressure instrumentation for wind tunnel models can be greatly reduced. Initially, low speed measurements of the mean pressure distribution on a delta wing were made in the Stanford Subsonic Windtunnel. Current work is taking place in the Research Test Facility (RTF) at NASA Ames, where measurements are being made on a three inch chord NACA 0012 airfoil. The technique has been employed successfully down to speeds as low as 8 m/s.

Project 5 - Prediction of wing maximum lift for preliminary design

The high lift characteristics of wings have important effects on aircraft noise, cost, and performance. The proposed research is aimed at improved understanding of the high lift flow regime for general lifting surfaces in the context of preliminary analysis and design. Computational models are being developed and used to examine the important inviscid and viscous phenomena that effect wing maximum lift, as well as the importance of three-dimensionality on the flow field. The ultimate goal is to develop an aerodynamic design tool that accurately models the effects of various design parameters on important performance criteria, especially maximum usable lift. This module will then be incorporated in a multi-disciplinary lifting surface optimization program for use in conceptual design of new airplanes.

Project 6 - Acoustic wave scattering through a turbulent mixing layer

Acoustic source location mapping techniques enable a detection of the noise source locations and provide a map of the acoustic "source" fields. Such maps can help reduce the noise of propulsion systems and aircraft. In open jet facilities, the noise produced by the system being tested propagates through a turbulent mixing layer before it is measured. Since tests are usually carried out on scaled models, it becomes necessary to measure high frequency sound sources (10 kHz or higher). The acoustic wave length is then comparable or smaller than the thickness of the turbulent mixing layer. For such high frequency waves, scattering phenomena becomes significant and simple refraction theories cannot predict the refraction corrections adequately. To overcome the aberration due to acoustic scattering at high frequencies, we need to quantitatively investigate this phenomena. Our approach is to numerically solve the subsonic jet flows using Large Eddy Simulation (LES), and study the sound scattering by generating incident sound waves due to a known source. Through this research, approximate descriptions of scattering can be validated which may allow more accurate source mapping techniques to be developed in future.

1.3 INSTITUTIONAL SUPPORT

Institutional support involves administrative, secretarial and technical salaries, travel, university equipment and services including communication, expendable supplies, computer services, engineering services, etc., and capital equipment. This support provides all of the basic services necessary for continuing operations of the Institute including its small-scale experimental and computational facilities, instrumentation and equipment and thereby supports all of the research and training activities summarized previously.

1.4 TRAINING ACTIVITIES

The training role of the Institute is accomplished through 6 units of coursework in acoustics offered by the Aero/Astro department including AA 201A (Fundamentals of Acoustics) and AA 201B (Topics in Aeroacoustics).

1.5 RESEARCH PARTICIPATION

The research programs summarized in Item 1.2 above will be undertaken by Stanford faculty, staff and graduate students within the Department of Aeronautics and Astronautics with the involvement of 4 Professors and 6 Ph.D. students. This group has experience in Aerodynamics and Acoustics and is familiar with NASA's wind tunnel and computational facilities. The strong collaboration between Stanford and Ames researchers which has been the hallmark of Joint Institute research in the past will be continued and enhanced in the coming year. The program activity at Ames will be coordinated by the Institute Associate Director for Center Affairs, Dr. C.A. Smith.

2. DETAILED PROGRAM DESCRIPTION

The research program proposed for the year, October 1, 1997 to September 30, 1998 is described in detail below. It has been discussed with the cognizant personnel at Ames and agreement has been reached on the general scope of the programs.

2.1 PROJECT 1 - ACTIVE FLOW CONTROL

Research participants: Prof. S. Rock, graduate student

Ames Technical Contact: Dr. C.A. Smith

2.1.1 *Introduction*

Controlled flight at high angles of attack provides increased maneuverability for fighter aircraft and increased lift during take off and landing. At these flight regimes, flow separation and vortex breakdown decrease the efficiency of conventional control surfaces when they are most needed to combat the onset of asymmetric flow. As a result of the inefficiency of the conventional control surfaces at these high angles of attack, alternate means to supplement the vehicle flight control are necessary. This research investigates the augmentation of aircraft flight control systems by the injection of a thin sheet of air tangentially to the fore-body of the vehicle. The method known as Fore-body Tangential Blowing (FTB) is proposed as an effective means of altering the flow over the fore-body of the vehicle (Ref. [1.1] and Ref. [1.2]). By using this method, the flow asymmetries are changed and consequently the aerodynamic loads are modified (Ref. [1.1] to Ref. [1.9]).

Static and dynamic experiments performed at the Department of Aeronautics and Astronautics at Stanford University under the NASA-JIAA program have shown that significant side force, roll and yaw moments as well as normal force and pitching moment (Ref. [1.5], Ref. [1.7] to Ref. [1.10]) can be generated using a small amount of blowing. This is important given that the implementation on a real aircraft would provide a limited amount of air. It has also been demonstrated that FTB could successfully be used to suppress wing rock and to roll the model to a desired bank angle. In addition, it was shown that asymmetric

FTB could provide the necessary aerodynamic force and moments for regulating aircraft yaw and roll and slewing to non-zero roll and yaw angles (Ref. [1.7], Ref. [1.8]).

During this past year, dynamic experiments have been conducted in the Stanford Low Speed Wind Tunnel using a wind tunnel model which is allowed two degrees of freedom: roll; and yaw (Ref. [1.7], Ref. [1.8]). This system provides a good approximation of the characteristics of the lateral-directional dynamics of an aircraft. The wind tunnel model underwent a modification to improve experimental reliability and currently consists of a cone-cylinder fuselage and a sharp leading-edge delta wing. While a vertical tail can be added, no movable surfaces are installed for control purposes. Movable flaps are currently being designed for the model and will be used in a related program to investigate their effectiveness with blowing present. The model is equipped with fore-body side slots through which blowing is applied. The amount of air injected is controlled by a closed loop control system employing specially designed servo valves and flow meters (Ref. [1.7] to Ref. [1.9]). A view of the wind tunnel test section with the unique two degrees of freedom apparatus and the wind tunnel model is shown in Figure 1.1.

The wind tunnel apparatus was used to increase the fundamental understanding of the underlying physics and experimental data measured, leading to the further development of an unsteady aerodynamic model that will support real-time control for commanding large-angle motion. This model builds upon the work done by Wong and Pedreiro (Ref. [1.7] to Ref. [1.9]) and describes the physics of the flowfield and the impact of blowing with a set of parameters that explicitly account for changes in the vortex flowfield as a function of the wind tunnel model's attitude and blowing.

2.1.2 Research Objectives

The overall objective of this research is to understand better the mechanisms through which tangential forebody blowing works and to further develop its application for lateral control of a wind tunnel model in two degrees of freedom. The ultimate purpose is to determine the feasibility of its use to control and/or improve the motion of an aircraft at high angles of attack.

2.1.3 Research Program

Experimental investigations have been conducted with a wind tunnel model with yaw and roll degrees of freedom. Static and dynamic measurements of the aerodynamics have been used to characterize the natural behavior of the system and the effect of blowing. A nonlinear mathematical extension of the current model of the system is being generated for use in the synthesis of control laws to provide the capability to command large roll and yaw angles, ϕ and γ respectively. Past work (Ref. [1.5] to Ref. [1.8]) has shown that blowing has a significant impact on the structure of the vortical flowfield, and can be har-

nessed to provide the control authority required to regulate aircraft roll and yaw and slew the aircraft to non-zero yaw and roll and angles. Furthermore, FTB is a nonlinear actuator is not represented well as an incremental control device because the aircraft stability derivatives depend nonlinearly on the level of blowing. The models generated by Wong and Pedreiro (Ref. [1.7] to Ref. [1.9]) lump the effects of FTB, using a small number of parameters to determine the dynamics of the vortical flowfield, prior to linearization. These models provide the necessary information required for real-time regulation about small angles and thus the basis for demonstrating that FTB augments the vehicle's conventional control systems. However, they are not capable of supporting real-time control for commanding large angle motion because of the corresponding increase in model complexity and detail required to support such a task. The current model can be extended and modified to present a more detailed picture of the aerodynamics to help predict the behavior of the moments and loads on the aircraft model as it undergoes large angle motion. Using FTB in real aircraft under realistic conditions requires further understanding of the underlying physics of such phenomena as vortex breakdown. This understanding can only be achieved through the development of a nonlinear model. Such a model would be accurate and detailed enough to provide reliable predictions for vehicle design and optimization.

2.1.4 Research Activities Completed During 1996-1997

Using the apparatus that was previously designed and built under the NASA-JIAA program, dynamic and static experiments were carried out to extend the current understanding of the aerodynamics of the phenomena and the use of blowing to control the roll-yaw motion of an aircraft at high angle of attack. The following results were achieved in the 1996-97 period.

(1) An investigation into possible modifications to the nose geometry was completed and a new forebody nosecone with a slightly rounded conical tip was fabricated to improve experimental repeatability. The previous nosecone was a sharp-pointed circular cone which exhibited a strong susceptibility to flow asymmetry. This asymmetry is in turn generated by the micro-asymmetries of the nose. Should small changes in the nose geometry occur, the required level of measurement precision would be compromised. This experimental observation has been made both at Stanford [Refs. 1.4, 1.8] and elsewhere [Refs. 1.11, 1.12]. Pedreiro [Ref. 1.8] overcame this problem by consistently "fixing" the nose throughout all the experiments. Research done on the subject suggests that changing the nose geometry to one with a chined cross-section [Refs. 1.13, 1.14] or a circular cone with a slightly-blunted tip would alleviate asymmetric flow and provide a more stable flow condition overall by reducing the flow's sensitivity to changes in the nose geometry [Refs. 1.4, 1.15-1.18]. The slightly-blunted tip was chosen as the new configuration after experiments in the wind tunnel confirmed measurements to be repeatable, without any loss of nonlinearity in the flowfield behavior.

(2) Progress has been made in increasing the current level of understanding of the aerodynamics of the phenomena and the effect of blowing. This progress has led to the development of an unsteady aerodynamic model that provides a more complete description of the coupling at high angle of attack between aircraft dynamic motion in roll, the aerodynamic forces and moments generated by the vortical flowfield and the effects of FTB. Currently, progress is being made on identifying the numerical values of the parameters for rolling motion only. The new model builds on work done by Wong and Pedreiro [Refs. 1.5-1.9] and reflects the results of recent experimental work done at Stanford and elsewhere [Refs. 1.33, 1.34].

This model accounts for the effect of the flowfield vortices on the aerodynamic loads and moments, associated with the widely different time scales encountered as well as the abrupt changes in the structure of the vortical flowfield as the aircraft assumes different attitudes. The effect of the vortices is dominated by the large time lags associated with the movement of the vortex burst point. The remainder of the flowfield behaves very nearly like potential flow and exhibits a very fast response to changes in aircraft attitude, reacting at the convection speed U_∞ . For our purposes, this can be considered to be instantaneous. Thus, the dynamic potential flow contribution is described as the instantaneous static potential flow contribution. The static potential flow contribution can be calculated and subtracted from the static total measured load and moments to determine the static vortical flow contribution. The potential contribution is very nearly linear and exhibit no discontinuities. The discontinuities or jumps found in the static total measurements are thus retained solely in the vortical contribution. This result is validated by flow-visualization experiments carried out at Stanford and elsewhere [Refs. 1.6-1.8, 1.34]. Their results characterize the discontinuities or "critical states" as changes in the static flow topology, such as vortex burst points crossing the trailing edge of the wing or a rearrangement of the vortices' location relative to each other and the aircraft. In this way, the nonlinear behavior of the aerodynamic loads and moments can be represented in the following manner:

$$A(\phi, \gamma, t) = A_{\text{static}}^{\text{potential}}(\phi, \gamma, \tau) + A_{\text{dynamic}}^{\text{vortical}}(\phi, \gamma, \tau) \quad (2.1)$$

where $A(\phi, \gamma, t)$ is an aerodynamic load or moment., the potenetial-static term is instantaneous, and the vortical-dynamic term is lagged.

Nonlinear indicial response methods [Refs. 1.19-1.34] were used as the mathematical basis for developing expressions for each of the terms in Equation 2.1. This method provides for the handling of multiple solutions such as bifurcations, jumps, and other nonlinear phenomena which typically occur in high angle-of-attack vortical flows. It is also flexible enough to accommodate the introduction of free non-motion variables such as blowing. This will allow for the development of an aerodynamic model that is more accurate than those developed by Wong and Pedreiro. With this method, the response to an arbitrary motion input is obtained by the superposition of responses to successive discrete steps. The essential difference between the linear and nonlinear cases is that the response of each step depends on the motion history in the nonlinear case, and not just on the instantaneous state of the system. In the absence of linearity, a remnant of the past must exist in the behavior of the indicial response because exact cancellation of past behavior can no longer be assumed, implying that the indicial response must be a functional. This methodology accounts for hysteresis and other nonlinearities exhibited by the plant.

With nonlinear indicial response methods, the terms in Equation 2.1 can be expanded in the following manner (using Myatt's notation [Refs. 1.30, 1.34]) to arrive at a model for no critical state crossings:

$$A_{\text{static}}^{\text{potential}}(\phi, \gamma, t) = A_s^p(\phi(t), \gamma(t)) + A_{\dot{\phi}}^p \dot{\phi}(t) + A_{\dot{\gamma}}^p \dot{\gamma}(t) \quad (2.2)$$

and

$$A_{\text{dynamic}}^{\text{vortical}}(\phi, \gamma, t) = A_s^v(t=0) + A_{s_{\text{lag}}}^v(t) + A_{\dot{\phi}_{\text{lag}}}^v(t) + A_{\dot{\gamma}_{\text{lag}}}^v(t) \quad (2.3)$$

where, in the Laplace domain,

$$A_{s_{\text{lag}}}^v(s) = \frac{A_s^v(\phi(s), \gamma(s))}{\beta s + 1} - \frac{A_s^v(\phi(t=0), \gamma(t=0))}{s(\beta s + 1)} + \frac{A_{s_{\text{lag}}}^v(t=0)}{s} \quad (2.4)$$

$$A_{\dot{\phi}_{\text{lag}}}^v(s) = \frac{\alpha_1 s + \alpha_0}{\beta s + 1} \dot{\theta}(s) \quad \theta = \phi, \gamma \quad (2.5)$$

The parameters for the model can be determined with regressive analysis techniques. If the motion is prescribed to a region where no critical states exist, i.e.; a region of equilibrium-flow stability, the parameters' values will remain constant and are unique. However, if the motion should involve crossing one

or more critical states, the parameters will change in value to reflect the changes in the topology of the vortical flowfield with each crossing, reflecting the different lag times of the movement of the vortex burst points [Ref. 1.34] within each region. There is also a transient associated with each critical state crossing which is unique but has a generalized form. It can be assumed to depend only on ϕ_c , γ_c and $\phi(\tau_c)$. In this case, the model for motion with a critical state crossing at $t = \tau_c$ is:

$$\begin{aligned} \tilde{A}(\phi, \gamma, t) = & A(t = 0) + [A_s^p(\phi(t), \gamma(t)) + A_\phi^p \dot{\phi}(t) + A_\gamma^p \dot{\gamma}(t)]_1 \\ & + [A_s^p(\phi(t), \gamma(t)) + A_\phi^p \dot{\phi}(t) + A_\gamma^p \dot{\gamma}(t)]_2 \\ & + [A_{s_{lag}}^v(t) + A_{\phi_{lag}}^v(t) + A_{\gamma_{lag}}^v(t)]_1 \\ & + [A_{s_{lag}}^v(t) + A_{\phi_{lag}}^v(t) + A_{\gamma_{lag}}^v(t)]_2 \\ & + \Delta A[\phi(\xi), \gamma(\xi); t, \tau_c] \end{aligned} \quad (2.6)$$

where the subscripts 1, 2 designate the regions of equilibrium-flow stability the motion passed through before and after the critical state crossing.

The effect of blowing on the vortical flowfield and the aerodynamic loads and moments can be considered to be similar to a critical state crossing. Flow visualization experiments and static and dynamic load measurements show that the response of the vortical flowfield structure to blowing is nonlinear and exhibits a time lag similar in form to that corresponding to the motion of the vortex burst points due to aircraft motion. Because the blowing takes place on the leeward side of the aircraft, and directly affects the vortical flowfield, its impact on the potential flow contribution terms can be assumed to be negligible. There is also a transient term which is a function of the direct reaction to the change in blowing jet momentum. This transient's effect on aerodynamic loads is highly dependent on aircraft attitude. As a result, the model for an aerodynamic load or moment with a change in blowing jet momentum at $t = \tau_\mu$ but no critical state crossings is:

$$\begin{aligned}
 \bar{A}(\phi, \gamma, C_\mu, t) = & A(t = 0) + A_s^p(\phi(t), \gamma(t)) + A_\phi^p \dot{\phi}(t) + A_\gamma^p \dot{\gamma}(t) \\
 & + A_{s_{lag}}^v(\phi(t), \gamma(t), C_{\mu_1}) + A_{\phi_{lag}}^v(\phi(t), \gamma(t), C_{\mu_1}) \\
 & + A_{\gamma_{lag}}^v(\phi(t), \gamma(t), C_{\mu_1}) + A_{s_{lag}}^v(\phi(t), \gamma(t), C_{\mu_2}) \\
 & + A_{\phi_{lag}}^v(\phi(t), \gamma(t), C_{\mu_2}) \\
 & + A_{\gamma_{lag}}^v(\phi(t), \gamma(t), C_{\mu_2}) + \Delta A[\phi(\xi), \gamma(\xi), C_\mu(\xi); t, \tau_\mu]
 \end{aligned}
 \tag{2.7}$$

where , $C_\mu \equiv \frac{\dot{m}_j V_j}{q_\infty S_{ref}}$ the jet momentum coefficient.

Current work is focused on refining this model and determining the parameters of the model for use in generating control laws that will allow for the commanding of large-angle motion.

2.1.5 Research Activities Proposed for 1997-1998

The following tasks are proposed to develop our understanding of the mechanisms through which tangential fore-body blowing works and explore the feasibility of its use to control and/or improve the motion of an aircraft at high angles of attack:

1. Continue developing the aerodynamic model to provide a more complete description of the coupling at high angle of attack between aircraft dynamic motion, the aerodynamic forces and moments generated by the vortical flow-field and the effects of FTB. The following steps are necessary to complete its development:

- Complete the survey of critical states using static data for moments and forces vs. roll and yaw angles. The critical states are "points" where equilibrium-flow stability is lost and marked by discontinuities in otherwise analytic relationships between moments and loads and aircraft attitude. This survey includes using flow visualization techniques to understand the processes of structural change at each critical state.
- Complete the determination of the vortical flow contribution parameters for each region of equilibrium-flow stability. This includes inducing simple harmonic motion in roll for a given yaw angle in regions between critical states, using a motor that will be attached to the model's sup-

port structure. The current wind tunnel apparatus has been designed to support a motor that will do this exact task. Regressive analysis techniques are applied to determine the significant onset parameters to create the NIR-based model of the aerodynamic loads and moments for motion proscribed to regions between critical states.

- The transition through critical states, as described by Equation 3, can be modeled by measuring the moments and loads as a prescribed “ramp-and-hold” motion is inputted to the model. The regressive techniques would again be applied to find the necessary parameters. This would provide an NIR-based model for motion over the desired range of roll and yaw angles.
- The parameters defining the effect of blowing will be found using regressive analysis on the dynamic measurements of the moments and loads as they respond to incremental changes in blowing. This will provide the model for the coupling between aircraft motion and blowing's impact.

(2) Experimentally demonstrate the utility of the aerodynamic model by using it to formulate control laws to command large roll and yaw angles at high angles of attack using FTB as the actuator.

2.1.6 Research activities planned beyond 1997-1998

- (1) Formulate control laws based on the aerodynamic model to allow for the capability to command large roll and yaw angles at high angle of attack using FTB as the actuator.
- (2) Experimentally demonstrate the capability to command large roll and yaw angles at high angle of attack using FTB in the wind tunnel.

2.1.7 References

- [1.1] Skow, A.M., Moore, W.A. and Lorincz, D.J., “Forebody Vortex Blowing - A Novel Concept to Enhance the Departure/Spin Recovery Characteristics of Fighter Aircraft,” AGARD Conference Proceedings 262, No. 24, May 1979, pp 24-1 to 24-17.
- [1.2] Celik, Z.Z. and Roberts, L., “Vortical Flow Control on a Slender Body at High Angles of Attack,” AIAA Paper 91-2868, 1991.
- [1.3] Font, G.I., “Tangential Fuselage Blowing on an Ogive Cylinder,” Stanford University Doctoral Thesis, Department of Aeronautics and Astronautics, June 1992.

- [1.4] Celik, Z.Z. and Roberts, L., "Vortical Flow Control on a Wing-Body Combination Using Tangential Blowing," AIAA Atmospheric Flight Mechanics Conference, August 10-12, 1992, Hilton Head Island, SC.
- [1.5] Celik, Z.Z., Pedreiro, N. and Roberts, L., "The Control of Wing Rock by Forebody Blowing," AIAA Paper 93-4386.
- [1.6] Celik, Z.Z., Pedreiro, N. and Roberts, L., "Dynamic Roll and Yaw Control by Tangential Forebody Blowing," AIAA Paper 94-1853.
- [1.7] Pedreiro, N., Rock, S.M., Celik, Z.Z. and Roberts, L., "Roll-Yaw Control at High Angle of Attack by Forebody Tangential Blowing," AIAA Paper 96-0773, January 1996.
- [1.8] Pedreiro, N., "Experiments in Aircraft Roll-Yaw Control Using Forebody Tangential Blowing," Stanford University Doctoral Thesis. Department of Aeronautics and Astronautics. To be published.
- [1.9] Wong, G.S., "Experiments in the Control of Wing Rock at High Angles of Attack Using Tangential Leading Edge Blowing," Ph.D. Dissertation, Aeronautics Department, Stanford University, 1991.
- [1.10] "The research and training activities for the Joint Institute for Aeronautics and Acoustics," Principal Investigator Professor Brian Cantwell, AERO No. 69-95, August, 1995.
- [1.11] Lowson, M. V. and Ponton, A. J. C., "Symmetric and Asymmetric Vortex Flows at High Angle of Attack," AIAA Paper 91-0276, Jan. 1991.
- [1.12] Alcorn, C. W., Croom, M. A., and Francis, M. S., "The X-31 Experience: Aerodynamic Impediments to Post-Stall Agility," AIAA Paper 95-0362, Jan. 1995.
- [1.13] Carr, P. C. and Gilbert, W. P., "Effects of Fuselage Forebody Geometry on Low-Speed Lateral-Directional Characteristics of Twin-Tail Fighter Model at High Angles of Attack," NASA Technical Paper 1592, 1979.
- [1.14] Brandon, J. M., "Low-Speed Wind-Tunnel Investigation of the Effect of Strakes and Nose Chines on Lateral-Directional Stability of a Fighter Configuration," NASA Technical Memorandum 87641, 1986.
- [1.15] Erickson, G. E. and Brandon, J. M., "Low-Speed Experimental Study of the Vortex Flow Effects of a Fighter Forebody having Unconventional Cross-Section," AIAA Paper 85-1798, 1985.
- [1.16] Hall, R. M., "Influence of Forebody Cross-Sectional Shape on Wing Vortex Burst Location," AIAA Paper 86-1835, 1986.
- [1.17] Roos, F. W. and Kegelmann, J. T., "Aerodynamic Characteristics of Three Generic Forebodies at High Angles of Attack," AIAA Paper 91-0275, Jan. 1991.

- [1.18] Ravi, R. and Mason, W. H., "Chine-Shaped Forebody Effects on Directional Stability at High- α ," *Journal of Aircraft*, Vol. 31, No. 3, pp. 480-487, May-June 1994
- [1.19] Tobak, M., Chapman, G.T. and Schiff, L.B., "Mathematical Modeling of the Aerodynamic Characteristics in Flight Dynamics," NASA Technical Memorandum 85880, January 1984.
- [1.20] Jenkins, J.E., "Relationships Among Nonlinear Aerodynamic Indicial Response Models, Oscillatory Motion Data and Stability Derivatives," AIAA Paper 89-3351-CONFERENCE PROCEEDINGS, 1989.
- [1.21] Jenkins, J. E., "The Nonlinear Indicial Response: vis-à-vis Roll-Rate Induced Camber Effects," AIAA Paper 94-3458, Aug. 1994.
- [1.22] Jenkins, J. E., "The Nonlinear Indicial Response: vis-à-vis Roll-Rate Induced Camber Effects," AIAA Paper 94-3458, Aug. 1994.
- [1.23] Jenkins, J. E., "Simplification of Nonlinear Indicial Response Models: Assessment for the Two-Dimensional Airfoil Case," *Journal of Aircraft*, Vol. 28, No. 2, pp. 131-138, Feb. 1991.
- [1.24] Reischel, P.H., "Development of a Nonlinear Indicial Model for Maneuvering Fighter Aircraft," AIAA Paper 96-0896, Jan. 1996.
- [1.25] Tobak, M., Chapman, G.T. and Schiff, L.B., "Mathematical Modeling of the Aerodynamic Characteristics in Flight Dynamics," NASA Technical Memorandum 85880, January 1984.
- [1.26] Tobak, M. and Schiff, L. B., "Aerodynamic Mathematical Modeling - Basic Concepts," AGARD LS-114, 1981.
- [1.27] Jenkins, J.E, Myatt J.H. and Hanff, E.S., "Body-Axis Rolling Motion Critical States of a 65-Degree Delta Wing," AIAA Paper 93-0621, Jan. 1993.
- [1.28] Hsia, A.H., Myatt J.H. and Jenkins, J.E, "Nonlinear and Unsteady Aerodynamic Responses of a Rolling 65-Degree Delta Wing," AIAA Paper 93-3682, Aug. 1993.
- [1.29] Hsia, A.H. and Jenkins, J.E, "Simplified Nonlinear Indicial Response Models of a Rolling 65 Degree Delta Wing," AIAA Paper 94-3493, Aug. 1994.
- [1.30] Jenkins, J.E. and Myatt, J.H., "Modeling Nonlinear Aerodynamic Loads for Aircraft Stability and Control Analysis," AGARD Report 789, pp. 13/1-13/10, Feb. 1993.
- [1.31] Jenkins, J. E., Private communication, July 1996.

- [1.32] Jenkins, J. E., "Aerodynamic Modeling for Flight Mechanics Analyses: Maintaining Fidelity in the Nonlinear Regime," 1st. Int. Conference on Nonlinear Problems in Aviation and Aerospace, ICNPAA 96 3.1.2, May 1996.
- [1.33] Internal Memo on NIR-based modeling of Aerodynamic Loads and Moments at High Angles of Attack, Dec. 1996.
- [1.34] Myatt, J.H., "Modeling the Rolling Moment on the 65-Degree Delta Wing for Rolling Motions at High Angles of Attack," Ph.D. Dissertation, Aeronautics Department, Stanford University, 1997.

2.1.8 *Figures*



Figure 1.1 Side View of Test Section

2.2 PROJECT 2 - LES AS A TOOL FOR STUDYING JET AERO-ACOUSTICS

Research Participants: Prof. S. Lele, graduate student

Ames Technical Contact: Dr. Nagi Mansour

2.2.1 *Introduction*

The prediction and control of aerodynamically generated noise is important to the design of quieter subsonic and supersonic aircraft. Current prediction methods primarily rely upon empirical formulae and the use some form of acoustic analogy (pioneered by Lighthill) for scaling data. The flexibility of such tools is limited by their empirical origins. With the recent advances in computational algorithms and computer hardware, a new generation of analysis tools can be developed which have a theoretical basis. For aero-acoustics, Large Eddy Simulation (LES) holds the promise of predicting the dominant features of noise radiated to the far-field by a flow such as a jet issuing from a nozzle.

Predicting the far-field noise via LES is far more challenging than an overall prediction of the near-field aerodynamic flow. Acoustic predictions are dependent upon a two-point space time correlation of flow quantities (Ref. [2.2]), a far more demanding test of the flow prediction's fidelity. By its nature, LES does not resolve all the dynamical scales of the flow being computed. The accuracy of LES results depends on the sub-grid scale models employed, which represent the range of scales not resolved in the simulation, and on the numerical algorithms being used (discretization scheme, time advancement, numerical boundary conditions, etc.). Recent developments in sub-grid scale models (Ref. [2.3]), while very attractive and promising for aerodynamic predictions, have not been tested from the viewpoint of far-field noise prediction. The present research is directed at extending the basic LES formulation to the prediction of far-field noise and testing how well such a method performs

During the last half of the preceeding year a new graduate student started work on this project. The equations used to simulate a statistically stationary coflowing jet flow were rederived. Numerical simulations were conducted (in collaboration with Dr. Robert Moser of the University of Illinois) using domain sizes significantly wider than in previous work, and a DNS of a higher Reynolds number jet was achieved. The data was analyzed to check if the statistical stationarity was being maintained. A broad range of measures were used and it was consistently observed that the flow was indeed stationary. Instantaneous flow structures were visualized (an example of which at two different times is depicted in Figure 2.1) and confirmed that statistically independent realizations were being achieved while maintaining stationarity. This validates the numerical approach to be used in the present study.

2.2.2 *Research Objective*

Under the premise that the dominant features of the far-field noise are associated with the dynamics of the energy containing range of scales in the near field aerodynamic flow, a Large Eddy Simulation which captures these energetic

scales can be expected to contain sufficient near field information to predict the far-field noise. The proposed research focuses on determining LES's efficacy in making acoustic predictions, considering the effects of sub-grid scale models.

Current sub-grid scale models which attempt to parameterize the influence of the unresolved scales have been developed to allow an overall statistical prediction of the aerodynamic near-field. The models are typically designed to provide correct energy transfer between the resolved and unresolved scales. While these models seem quite promising, their impact on far-field acoustics has not been considered. Since the far-field noise is a very small by-product of the flow, it is necessary to ensure that the sub-grid models do not behave as a low order (and hence efficient), but spurious source of sound. As Crighton (Ref. [2.1]) points out, sources of this type may be introduced via discretization errors and numerical boundary conditions, etc. It is necessary to examine sub-grid scale models in this context. Furthermore, for practical LES applications, the sub-grid scale energy may be as much as 10-30% of the resolved energy and this may require that closer attention be paid to the acoustic sources implied by such models.

2.2.3 Research program

We propose to carry out a program of research aimed at applying the LES methodology from the point of view of far-field noise prediction. The fidelity of LES in predicting the unsteady flow and acoustic sources will be judged by making extensive comparisons with a Direct Numerical Simulation (DNS) of the same flow configuration. For this reason, the first phase is a direct simulation of turbulent flow in simple geometries.

A formulation capable of yielding a stationary turbulent jet flow while maintaining the efficiency and accuracy of spectral methods was developed. The method is an extension of Spalart's method (Ref. [2.5]) for simulating boundary layers, to the case of a co-flowing jet or wake. The results resemble those obtained by Spalart (Ref. [2.6]) for the sink-flow boundary layer. The method involves incorporating the slow spatial growth effects via a decomposition of the variables according to their multiple spatial scales and a suitable coordinate transformation (Ref. [2.7]). The derivation is more rigorous than the boundary layer analysis, due to the simplification introduced by explicitly considering the small deficit limit.

The result of the formulation is a modified set of equations consisting of the Navier Stokes equations with an additional set of small growth terms. The implied flow field is homogeneous in both the stream-wise and span-wise directions, and was therefore implemented in the spectral code used by Rogers and Moser (Ref. [2.4]). Testing of the code's ability to maintain a stationary flow was carried out, and a preliminary Direct Numerical Simulation was completed.

2.2.4 Research activities proposed for 1997-1998

The DNS will first be extended to collect the data necessary to make acoustic predictions. This will involve continuing the computation from the final condition previously reached and computing the two point space-time correlations necessary for acoustic predictions. Concurrently, the code will be modified to perform the Large Eddy Simulation of the same flow field. The LES calculations will begin with the simplest Smagorinsky type sub-grid scale eddy viscosity models. Comparisons of the LES and DNS results will begin with one point statistics and move on to the two-point correlations mentioned previously. If the LES and DNS results compare well at this level it may be concluded that the dominant acoustic sources have been effectively modeled (at least in the context of an acoustic analogy).

2.2.5 Research Activities Planned Beyond 1997-1998

Once the initial tests of LES's fidelity have been carried out using a simple sub-grid scale model, the direct impact of different sub-grid scale models will be examined. This will require a study of the space time correlation of the model sub-grid stresses and the resolved stresses in the DNS database. It is expected that this will involve reintegrating the DNS data from the coarsely spaced times available as restart files. When the sub-grid scale energy is non-negligible these correlations may provide information about how much noise is radiated by the unresolved sub-grid scale motions and how effectively it is captured by sub-grid models employed in the calculations. Once the appropriateness of LES has been established, the prediction of far-field noise which uses the near-field unsteady flow calculated via LES can be applied to flows of engineering interest. LES calculations of an experimental flow for which detailed flow and noise data exist may be the next logical step in this direction.

2.2.6 References

- [2.1] Crighton, D.G., "Goals for computational aeroacoustics, in *Computational Acoustics: Algorithms and Applications*," Elsevier Science Publishers B. V., IMACS, 1988
- [2.2] Lighthill, M. J., "On sound generated aerodynamically (I). General Theory," *Pro. Roy. Soc. (A)*, 1952, 211, 564-587
- [2.3] Ghosal, S. and Lund, T. S. and Moin P., "A dynamic localization model for large eddy simulation of turbulent flows," *Center for Turbulence Research Manuscript*, 1992, 139
- [2.4] Moser, R. D. and Rogers, M. M., "Direct simulation of a self-similar plane wake," *NASA TM 108815*, 1994
- [2.5] Spalart, P. R., "Numerical study of sink flow boundary layers," *J. Fluid Mech*, 1986, 172, 306-328

- [2.6] Spalart, P. R., "Direct simulation of a turbulent boundary layer up to $R_q=1410$," J. Fluid Mech, 1988, 187, 61-98
- [2.7] Timson, S. F. and Lele, S. K. and Moser, R. D., "A method for obtaining a statistically stationary turbulent free shear flow," Center for Turbulence Research Annual Research Briefs, 1994, 365-371

2.2.7 *Figures*

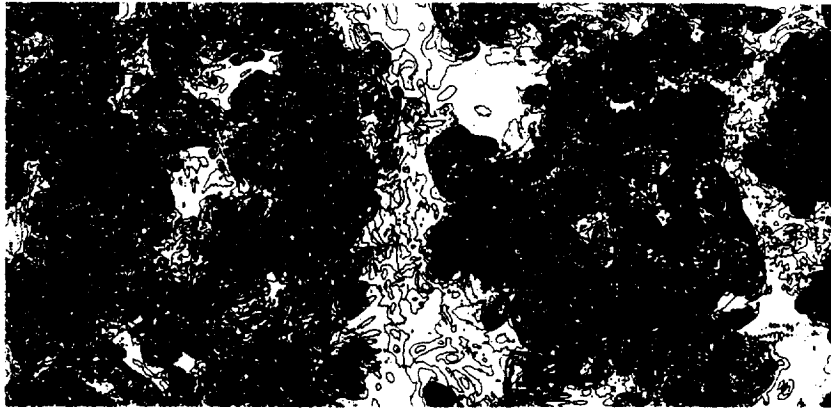


Figure 2.1 Instantaneous Vorticity Magnitude Field in the Center Plane at a Non-Dimensional Time of 1.13

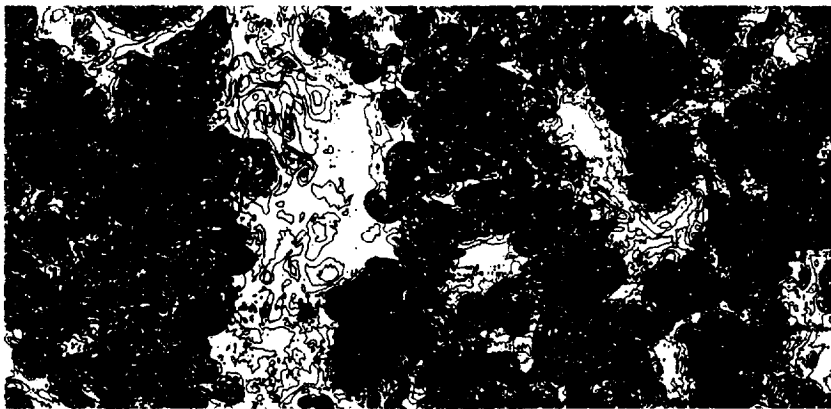


Figure 2.2 Instantaneous Vorticity Magnitude Field in the Center Plane at a Non-Dimensional Time of 4.01

2.3 PROJECT 3 - RESEARCH ON A LIFTING WING-FLAP CONFIGURATION

Research Participants: Prof. B. Cantwell, graduate student

Ames Technical Contact: Dr. L. Olsen, Dr. J. Ross

2.3.1 *Introduction*

Experiments by Kendall and his co-workers (Ref. [3.1], Ref. [3.3] and Ref. [3.4]) and Grosche et al. (Ref. [3.2]), using a directional microphone to measure noise distributions on a wing with semi-span flap model, found that the noise generation was highly localized. The most active locations were the wing tip and leading edge, the trailing corner and trailing edge of the flap, and the gap separating adjacent flap elements. Areas of attached turbulent flow were inconsequential noise sources. In experiments with individual flaps, vortex roll-up was postulated as the major reason for noise generation. When the vortex strength was reduced, the noise intensity decreased. Kendall observed that the major part of the noise generation was caused by the gap between two differentially deflected flaps. Kendall also argued that the trailing edge noise did not play an important role.

Recent experiments (Ref. [3.5] and Ref. [3.6]) on a wing-half span flap configuration by NASA investigators in the Ames 7 x 10 tunnel included a collaborative program of noise measurements by Boeing researchers. As with Kendall's findings, the flap edge was clearly identified as the major source of noise. Various edge treatments were tested in an attempt to reduce noise. Large differences in the effectiveness of noise reduction were observed depending on the particular choice of flap edge treatment.

A Fluid dynamic description of noise generation from a lifting surface is extremely complex: confluent turbulent boundary layers and vortex sheets roll over edges producing large surface pressure fluctuations. The flow involves a wide range of length scales, high local shearing stress and intense turbulence activity over the lifting surface. A better understanding of these processes is needed for the development of effective methods for airframe noise reduction.

2.3.2 *Research objective*

The objective of this research is to understand the flow mechanisms responsible for noise generation by a wing and trailing edge flap combination. A NACA 63-215 Mod B airfoil section with a Fowler Flap has been selected for flow measurements in the areas of high noise generation. For comparison purposes, the same model geometry used in the Ames 7 x 10 experiments will be used in computations by Stanford and Ames investigators and in small scale experiments at Stanford.

2.3.3 Research program

An identical model geometry was selected for experiments both at Stanford and NASA-Ames and for the computational work. The model is designed with two interchangeable middle parts to allow for instrumentation suited to either the wind tunnel or the water channel environments. For the wind tunnel experiments, the middle section is instrumented with pressure tapings and embedded pressure sensors. In this configuration, the model contains 35 pressure tapings on the main section and three pressure sensors on the side of the semi-span flap. For the water channel experiments, the middle section is replaced with a geometrically identical section containing dye ports instead of pressure tapings. The main wing has 12 dye ports; four on the top surface, seven on the bottom and one at the edge of the cove section. Of three dye holes on the flap, one is located on the top surface while the other two will be used to inject dye on the lower surface.

Initial measurements made in the Stanford tunnel with an uncalibrated hot-wire indicated that it would be necessary to improve the flow quality, add speed feedback, and install temperature regulation equipment. To this end the tunnel was dissassembled and variety of modifications were made. Temperature control was accomplished by adding a heat exchanger just before the first turning vanes downstream of the fan. A closed loop controller adjusting the chilled water supply to the heat-exchanger is able to maintain the temperature within 0.1C of the set-point. In the settling chamber, before the contraction to the test-section, the original screens were removed and replaced with a greater number having a slightly larger open area. A section of honeycomb was also added upstream of the screens for directionalization purposes. With these modifications it was found that the RMS turbulence level dropped from 3-5% to just below 1% although flow uniformity in the test-section cross-section decreased slightly. A controller for feedback on the tunnel speed has been obtained, but will not be installed until there is a break in the experiment schedule.

During the process of reinstalling the model in the test-section and making some preliminary surface pressure measurements - it was discovered that the free end of the cantilevered flap was undergoing high-frequency vibration. As the flap edge is a crucial measurement area, it was deemed essential that this problem should be resolved. A third mounting arrangement was constructed whereby a rigid rod penetrates the length of the flap and securely attaches to the micro-positioner fixed to the side-wall. With this new arrangement, flap-edge vibration is no longer visible at all but the highest flow speeds.

A computer controlled three-axis traverse was installed which will allow positioning to 0.001" not only in a single plane, but over the length of the test section as well.

A new data acquisition system has been installed that will allow for the simultaneously sampled, high resolution, high frequency measurements that multi-channel hotwire measurements require. Several of the data acquisition programs have been ported to this new platform.

2.3.4 Research activity proposed for 1997-1998

Three sets of measurements are planned for the 1997-1998 academic year. Using existing equipment, a 5-hole probe survey will be completed in the near wake region of the flap. Surface pressure transducers will be embedded in the model, and the pressure field will be characterized over the full range of position of the flap. A new hotwire data-acquisition system will be obtained and used to characterize the 2-dimensional turbulence field in the near wake region and the flap-edge. It is expected that computational results will be obtained simultaneously with the experiments. Comparisons between the two data sets will be possible in real time, which should provide a very natural way to determine in which direction the experiments/computation should evolve. A new smoke generation/laser light sheet system will be used to further examine the issue of separation over the main element.

At some point during the year, the problem of flow non-uniformity in the test-section will be revisited. It is expected that the addition of more screens - either in the settling chamber, or just before the last turning vanes should greatly alleviate this problem. If this should not prove to be the case, smoke-wand visualization will be used to inspect the flow in the diffuser to check for separation.

2.3.5 Research activity planned beyond 1997-1998

Eventually we would like to make use of the Luminescent Paint technique discussed in section 2.4. We hope to make pressure sensitive paint measurements on the wing to provide a more complete picture of the mean pressure field on the wing at various flap settings. A concurrent effort is underway to compute the same flow measured experimentally in these tests. It is the ultimate intent of this research to compare the results from these two data-sets and from recent acoustical surveys done at Ames to look for correlations that would suggest a tie between the noise generated, and the underlying physics of the local flow-field. This knowledge would have application as a design tool to quantify the effect that the flap edge vortex has on noise generation.

2.3.6 References

- [3.1] Kendall, J.M., "Airframe Noise Measurements by Acoustic Imaging," AIAA 77-0055, January 1977.
- [3.2] Grosche, F.R., Stiewitt, H and Binder, B., "Acoustic Wind Tunnel Measurements with a Highly Directional Microphone," AIAA Journal, Vol. 15, No. 11, pp.1590-1596, November 1977.
- [3.3] Kendall, J.M., "Measurements of Noise Produced by Flow Past Lifting Surfaces," AIAA 78-0239, January 1978.

- [3.4] Kendall, J.M. and Ahtye, W.H., "Noise Generation by a Lifting Wing/Flap Combination at Reynolds Numbers to 2.8×10^6 ," AIAA 80-0035, January 1980.
- [3.5] Storm, B. and Ross, J., "Flap Edge Post-Test Debrief," NASA TR-762, 1995.
- [3.6] Takahashi, T., "Wake Survey Experiments of NACA 63-215 Mod B Airfoil," 1995.

2.3.7 *Figures*



Figure 3.1 Semi-span Flap Model in Test-section

2.4 PROJECT 4 - LUMINESCENT PAINT FOR AERODYNAMIC MEASUREMENT

Research Participants: Prof. B. Cantwell, Graduate Student

Ames Technical Contacts: Dr. R. Mehta, Dr. S. Davis

2.4.1 *Introduction*

Pressure sensitive paints (PSP's) are now used routinely in industry and academia for measuring surface pressures on wind tunnel models at transonic and supersonic Mach numbers. This measurement technique utilizes a surface coating containing luminescent materials, the brightness of which varies with the local oxygen partial pressure on the surface. In current practice, a wind tunnel model is coated with the PSP, which is then illuminated with light of an appropriate wavelength to excite the material. The illuminated model is imaged with a digital CCD camera during the wind tunnel test. The images are then computer-processed in order to obtain a map of the surface pressure distribution. The relationship between surface brightness and pressure is generally determined by calibrating the paint (in situ) using a few pressure taps on the model. PSP's have some important advantages over pressure taps which are typically used for surface pressure measurements on wind tunnel models. First, pressure measurements using PSP's provide a complete pressure map over the entire model surface. Second, measurements using PSP's require less total time and effort as compared to those using pressure taps alone.

2.4.2 *Research Objective*

Low-speed ($M < 0.1-0.2$) aerodynamic testing is becoming increasingly relevant. For instance, complex multi-element systems are being designed and tested for subsonic and supersonic transports for the take-off and landing phases which must be extensively studied and tested in wind tunnel experiments. Experiments using the PSP technique have the potential of providing the surface pressure profile over an entire model in a relatively fast and inexpensive manner. However, one limitation of PSP's is that the brightness change of the paint is inversely proportional to the local pressure change (relative to ambient conditions). Therefore, detection of differences in brightness, which relate to differences in pressure on the model, become increasingly difficult as the flow speed is reduced. Eventually, as flow speed is lowered, measurement equipment noise will dominate the measurement signal. A second limitation is that the vast majority of these paints display an Arrhenius type sensitivity to temperature. Changes in the temperature profile about a wind tunnel model can therefore alter the calibrated pressure response. This phenomenon becomes more predominant at lower flow speeds, as the pressure induced response becomes more negligible. The overall objective of this research program is to apply the luminescent paint technology to the study of basic fluid physics problems, especially at low subsonic speeds. Issues regarding the limitations discussed at low speeds will be studied in an effort to improve the ability to utilize PSP's for measuring low-speed flows.

2.4.3 Research program

Research continued on low speed flows, following that conducted by Shimbo (Ref. [4.1]). The Research Test Facility (RTF) at the NASA Ames Research Center was utilized for all studies. The RTF houses a 12"x12"x18" wind-tunnel. All investigations were carried out in this tunnel on a subscale (3.0" chord) 0012 NACA series airfoil model. PSP (PtTFPP/FIB7), developed at the University of Washington, is being used for tests on this model. This paint has a slightly lower temperature sensitivity and it also photodegrades at a lower rate as compared to other PSP's. A 14-bit CCD camera interfaced to a Pentium PC is being used to collect images from the model. Images are then transferred to a SGI workstation, where data reduction takes place utilizing the pressure paint software package "Greenboot", developed by NASA Ames and McDonnell Douglas. Studies this year concentrated on low speed flow characteristics of the 0012 airfoil. Flow speeds from 10 to 50 m/s ($1 < |C_p| < 3$) were used to determine the paint pressure response characteristics on the wing. At first, data indicated that pressure changes on the order of 100 Pa could be detected at flow speeds down to 30 m/s (Ref. [4.2]). Below 30 m/s, signal to noise ratios were too low to identify any pressure changes on the model. Recent work has found that camera and lighting source stabilization techniques together with improved paint application processes can greatly improve the ability to measure pressure changes at lower speeds. Recent tests have shown characteristic pressure response on the 0012 wing at flow speeds down to 8 m/s. In addition, separation bubble phenomena are believed to have been observed using PSP's at approximately 15 m/s. There appears to be some temperature related effects on the measurements, which have yet to be determined and quantified.

2.4.4 Research activities planned for 1997-1998

Work will continue on the 0012 airfoil at the RTF. Response to pressures at the low flow speeds (8 - 50 m/s) will be fully quantified. Temperature effects will be fully investigated. In addition, flow phenomenon such as separation and transition will be examined. Future work will also include use of vortex generation devices on the wing to affect stall characteristics.

Multi-element airfoils will be utilized for PSP measurements in both the RTF and Stanford wind-tunnel. Measurements using PSP's on multiple element airfoils are challenging. First, reflected light from one area of the airfoil can interfere the a second area on the body (deemed "self-illumination"). Second, in order to obtain a complete image, multiple cameras and lighting systems must be used. The translation of data from two geometric perspectives into one three dimensional pressure grid is a problem requiring further research.

In order to fully understand the response characteristics of the PtTFPP/FIB PSP being used, as well as other potential paints, coupon paint samples must be evaluated over a range of pressure and temperature conditions. It is planned to conduct coupon sample testing in a pressure chamber currently undergoing

integration at NASA Ames. Use of this chamber will also allow characterization of the paint time-response as well as photo-degradation effects. Knowledge of PSP behavior due to state and time-related effects is vital in the ultimate determination of paint response and limits at low-speeds.

2.4.5 Research activities planned for beyond 1997-1998

Measurements using PSP's at low flow speeds will prove beneficial to the aerospace as well as the automotive industry. Ultimately the findings of this current research will be used to provide pressure maps for complex aerodynamic bodies at low flow speeds.

2.4.6 References

- [4.1] Shimbo, Y., Mehta, R.D. and Cantwell, B.J., "Application of the pressure sensitive paint technique to steady and unsteady flow," JIAA TR 115, June 1996.
- [4.2] Brown, O., Mehta, R.D. and Cantwell, B.J., "Low-Speed Flow Measurements at NASA Ames using Pressure Sensitive Paint," 1997 Pressure Sensitive Paint Conference, May 1997.

2.4.7 *Figures*



Figure 4.1 PSP measurement setup at the NASA AMES Research Test Facility.

2.5 PROJECT 5 - PREDICTION OF WING MAXIMUM LIFT FOR PRELIMINARY DESIGN

Research Participants: Prof. I. Kroo, Graduate Student

Ames Technical Contact: Dr. L. Olsen, Dr. S. Smith

2.5.1 Introduction

Traditional aircraft selection methods, such as parametric studies and summary charts, allow the designer to pick the "best" design based on variation of a limited number of parameters. This best design is, however, still sensitive to many other variables that were not examined in the original parametric studies. The true optimum design can often only be found when all of the available design parameters are varied simultaneously. This is especially true when exploring new, unconventional configurations. Recent advances in the field of multidisciplinary optimization make this approach promising for use in the conceptual design of new aircraft. The major benefit of computerized optimization for aircraft design is the ability to perform these trade studies with many more parameters and much greater speed than can be accomplished through traditional methods.

While numerical optimization provides the benefit of a more thorough exploration of the design space, it also presents new challenges for the modeling of the aircraft throughout that space. The analysis routines that provide objective and gradient information to the optimizer must be accurate, robust, and fast. In order to best meet their objectives, optimizers tend to push designs to their limits. This can result in exploitation of weaknesses in the aircraft model, producing designs that are not practical or feasible. The analyses must be sensitive to a wide range of parameters in order to fully explore the available design space, and they must also be simple enough to run quickly since the optimization process may require thousands of function evaluations. There is a fundamental trade-off between accuracy and speed that must be properly made in order to formulate analysis techniques that make numerical optimization a practical tool for use in conceptual and preliminary design.

Vortex-lattice codes are the primary aerodynamic analysis tools used in many multi-disciplinary design programs, including the one that this research seeks to improve on. In these methods, maximum lift, compressibility drag, and viscous effects are computed using simple sweep theory along with 2-d airfoil data and empirical corrections based on the computed lift distributions. This works well for the thin, high aspect ratio wings of most conventional transport aircraft, but may not adequately address the effects of large changes in sweep, taper, and thickness of some unconventional configurations. An aerodynamic design tool that properly handles important 3-d effects would enhance the capability of multi-disciplinary aircraft design programs to fully explore the available design space.

Wing maximum lift is one of the areas that has been most difficult to model accurately for optimization. It has also been shown to be a very important parameter for choosing optimum wing planforms (Ref. [5.1]), due to significant effects on aircraft noise, cost and performance. The trade-off between high sweep for low drag at high Mach numbers and low sweep for good low speed performance and handling qualities is of fundamental importance. Unconventional aircraft designs, like the McDonnell-Douglas Blended Wing-Body configuration, may also have wings with sharp changes in sweep and thickness at various span-wise stations that could have significant effects on the high lift aerodynamics of the wing. However, current methods used to evaluate wing maximum lift in conceptual design phases may not be sufficient to accurately model the effects of sweep and geometric irregularities.

The research program described here seeks to improve on the current method in many of the key areas described above. Inviscid criteria that can be used to identify the maximum usable lift condition have been identified. These include the appearance of sonic conditions at the pressure peak and a maximum allowable change in pressure coefficient over any given section. A 3-d lifting surface code has been developed to provide surface pressures and velocities, as well as efficiently computing stability derivatives and control surface effectiveness for single- or multi-element configurations. This code uses sufficiently simple geometric definitions and singularity distributions to provide analytic gradient information for the numerical optimization routine, while still allowing analysis and design of a wide range of aerodynamic shapes. A modified 2-d integral boundary layer method that includes the important local effects of sweep and taper has been identified and is currently being integrated with the 3-d panel code. The analysis codes described in this report have also been used extensively during the past several months as a key component of the design and testing program of the 17 foot span, remotely piloted Blended Wing-Body Flight Control Testbed.

2.5.2 Research Objectives

Some optimization results using the current analyses have revealed problems due to inadequately modeling the impact of large spanwise geometric variations on aerodynamic performance and controllability. The sweep, thickness, and planform variations of the Blended Wing-Body have been especially troublesome for the current method, suggesting that an improved set of analyses is needed to properly handle unconventional configurations. Usable maximum lift is a major constraint that limits the sweep of a wing. With optimization results favoring larger sweeps than would be expected, a better assessment of the penalties sweep will impose on maximum lift capabilities is needed. Additionally, attempts to model thickness effects while still using the current Weissinger aerodynamic code have produced unsatisfactory results for large thickness-to-chord wing sections. Therefore, we propose to continue a program of research to 1) determine the effects of wing sweep and other geometric properties on

maximum usable section lift and 2) develop an aerodynamic model and optimization routine suitable for use in the early stages of conceptual design of airplanes.

2.5.3 Research program

The starting point for the research described here is a lifting surface design program, WingMOD, developed by Sean Wakayama at Stanford University (Ref. [5.7]), that is currently being used at McDonnell-Douglas. To determine wing maximum usable lift, this method uses a critical section analysis which compares local section lift coefficients, calculated from a Weissinger vortex lattice method, with estimates of 2-d maximum section lift coefficients based on empirical data. Flaps are simulated by increasing wing incidences in the Weissinger model, applying an increment in $c_{l,max}$ due to flap deflection on the flapped portion of the wing, and increasing the assumed $c_{l,max}$ due to induced camber on the sections near the flap edge. Finally, the maximum 2-d section lift coefficients are reduced by a factor of $\cos\Lambda$ as an empirical correction for the effect of sweep on the pressure distribution. The wing is then assumed to be at its maximum usable lift when any section c_l reaches some fraction of its local $c_{l,max}$. The assumption of a $\cos\Lambda$ variation in $c_{l,max}$, as opposed to the simple sweep theory (Ref. [5.2]) assumption that $c_{l,max}$ decreases with $\cos^2\Lambda$, is based on experimental observations (Ref. [5.3]). It has been observed that proper placement of fences and vortex generators on swept wings can yield $c_{l,max}$ values that closely approach the 2-d unswept values (Ref. [5.4], Ref. [5.5]).

The critical section method may be justified for unswept wings, but its validity is suspect for wings with significant sweep for several reasons. The method assumes acquisition of 2-d $c_{l,max}$ by placement of boundary layer control devices without actually specifying the location or geometry of such devices. Also, wing sweep changes the shape of the pressure distribution at fixed total lift, increasing the magnitude of the leading edge pressure peak. Finally, the existence of transverse pressure gradients along a swept wing induces boundary layer flow in the span-wise direction. This span-wise flow increases the length over which the boundary layer develops, resulting in a weaker boundary layer toward the wing tip. In certain codes used for high lift design at Boeing (Ref. [5.6]), 3-d panel codes are coupled with 2-d boundary layer codes, thus capturing the correct 3-d pressure distribution, but still neglecting 3-d boundary layer effects. Computational models are being developed and used to examine various inviscid and viscous phenomena that effect the maximum usable lift of wing designs. The results of this study will be used to formulate an improved algorithm for optimization of lifting surfaces in preliminary design methods.

There are several other aspects of the current aerodynamic analysis that need improvement, and are being addressed in the current research program. Boundary layer quantities are computed using a 2-d airfoil code, matching the lift coefficient for each local station as computed by the vortex-lattice code.

Although corrections can be made for simple sweep, other 3-d aerodynamic effects that alter the pressure distributions (and therefore the boundary layer) are not properly accounted for. As a result, the 2-d estimates of section $c_{l,max}$ do not give a true sense of the wing maximum lift. Additionally, the effects of the boundary layer on the lift distribution and the viscous pressure drag are not properly modeled. The current analysis uses a crest critical Mach number model to predict compressibility drag that is based on the Mach number perpendicular to the wing section, the section t/c , and the local lift coefficient. This method relies on flight test data from several commercial transports (Ref. [5.8]), and may not translate accurately to unconventional configurations. One of the most important aspects of this research program is the development of a 3-d aerodynamic code to address these shortcomings of the current method.

During the first year of this research program, studies were initiated in two main areas. The first study focused on transverse boundary layer development, and the second on the effect of sweep on the inviscid pressure distribution for a wing of fixed total lift. The transverse boundary layer development is studied using a 3-d, incompressible Navier-Stokes code (INS-3D) to compute the flow properties along a segment of a swept wing. The wing segment runs all the way to the boundaries of the computational grid, eliminating the need for wing tips or roots. The idea behind modeling only a section of the wing is to use the available computational resources as efficiently as possible. By eliminating the wing tip and root, more grid points can be concentrated along the section of the wing where the boundary layer development is examined. Additionally, the grid is relatively easy to generate, allowing fast and efficient parametric studies varying sweep and airfoil sections. The key element of the model is then applying appropriate boundary conditions to simulate the actual flow over a section of a swept wing with or without a boundary layer control device.

Several cases were run using the model described above with some promising results. The code was able to converge using the unconventional boundary conditions and evidence of a growing span-wise boundary layer can be seen in the flow solution. Substantial span-wise flow and some outboard flow reversal could also be seen using particle traces. Also, a code to compute boundary layer characteristics in the chord-wise and span-wise directions was developed and tested on some of the preliminary test cases.

In addition to the viscous phenomenon discussed above, an inviscid effect commonly observed to cause separation is sonic flow at the pressure peak (Ref. [5.3]). The lift coefficient at which this sonic flow condition is reached is affected by sweep for a couple of reasons. Since the normal component of Mach number is reduced by sweep, the increment in velocity required to bring the flow to sonic conditions is increased. However, from simple sweep theory, for a fixed free-stream lift coefficient the effective section lift coefficient on a swept section must increase by a factor of $1/\cos^2\Lambda$. This results in a higher magnitude pressure peak with increasing sweep for a given total wing lift. Since these two effects of sweep are competing, it is of interest to develop a model examining

the effect of sweep on maximum lift limited by sonic velocity at the pressure peak. A method of computing the lift coefficient at which sonic flow appears for a given configuration was developed using a 2-d panel code, and is described in detail in last year's report. During the past year, this method has been adapted to the new 3-d aerodynamic code described below.

With this preliminary work done on modeling some boundary layer flow and maximum lift criteria, the next step in this research program was to develop a simple 3-d aerodynamic code to use as a basis for the wing design/optimization program. The criteria for this model were that it be simple and fast while giving a fairly accurate 3-d pressure distribution over general wing geometries. Simplicity in both defining the geometry and solving for the 3-d flow were considered important features since the code will be used in a multi-disciplinary optimization environment. The geometric definition and singularity distributions were chosen to allow analytic computation of gradient information for the optimizer, as opposed to the finite difference gradients that would be required for a more complex aerodynamic code. This is a key feature for optimization due to the large difference in speed of execution between the two options. In order to compute the necessary gradient information required to do an optimization step in n design variables, a code with analytic gradient information requires only one flow solution, while a code requiring finite difference gradient evaluation requires $2n$ flow solutions. Since the primary focus of this research is low speed, high lift flight, a linear aerodynamic panel method was chosen.

As a preliminary step in the design of the panel code, several 2-d experiments were performed to examine ways of simplifying the 3-d method. These 2-d experiments proved very helpful, and eventually led to the selection of the algorithm for the 3-d aerodynamic model. Some of these experiments, as well as the problems encountered by the original 3-d code, are described in detail in last year's report. The details of the current 3-d code are presented here.

Constant strength quadrilateral source panels are placed on the surface of the wing, with vortex panels on the wing's mean surface. The geometry is illustrated in Figure 5.1 and Figure 5.2. The component of the distributed vortex sheet in the direction normal to the free-stream flow (the y direction) is set as constant at each given span-wise station. The component of vorticity in the x direction on the vortex panels and in the wake is computed according to Helmholtz second vortex theorem. The boundary conditions for this formulation are tangent flow conditions at each surface control point and Kutta conditions at the trailing edge panel control points at each span-wise station. There are several points that make the mean surface method an attractive alternative to a source-doublet surface panel code. The number of AIC calculations required is considerably lower, and the lift and induced drag can be computed using vortex theorems rather than integration of surface pressures. Another possible benefit is the improved condition of the system of equations to be solved for wings with very thin trailing edges. Doublet codes have historically shown conditioning

problems when the upper and lower surface doublet panels get too close together, while the mean surface vortex has no upper or lower surface counterpart.

The original 3-d code gave good results for some simple cases, but did not work satisfactorily for all general geometries. Many improvements have been made in the past year, and the code has given very good results for a wide range of test cases. Analytic influence coefficients have been computed for general quadrilateral panel shapes, allowing fast flow solutions and gradient calculations, and the geometry routine has been modified to allow efficient grid definition for any general body. The discontinuous vorticity distribution of this method results in the appearance of discrete vortices at the spanwise panel junctions, inherently resulting in poor lateral velocity matching for thin wings. However, a method for distributing these vortices, to more accurately model the actual physical flow conditions, has been developed and tested with excellent results. The velocity and pressure distributions computed using this code have been shown to match those of the higher order panel code A502 to within a few percent for a wide range of configurations. (Figure 5.3)

The current 3-d aerodynamic code is equipped to provide much of the important information necessary in a multi-disciplinary design environment. Full 3-d lift, induced drag, and velocity distributions can be computed for single- or multi-element configurations. Stability derivatives and control surface effectiveness are computed quickly and efficiently. Linear estimates of control authority for hinged surfaces can be computed by simply rotating the boundary conditions at control surface panels, affecting only the right hand side of the flow equation and eliminating the need to recompute aerodynamic influence coefficients. A 3-d implementation of the Prandtl-Glauert compressibility correction allows accurate lift and compressibility drag estimates at moderate airspeeds. Most significantly, this code has already proven to be a valuable design tool.

During the past several months, the design code developed under this program has been used extensively in the design and testing of the Blended Wing-Body 17 ft. Flight Control Testbed (BWB-17). The BWB is a new concept airplane currently under development by McDonnell-Douglas, NASA, and a group of universities including Stanford. As its name implies, the BWB configuration combines the wing and fuselage into one continuous lifting body for improved structural and aerodynamic efficiency over conventional transports. The BWB-17 is a dynamically-scaled, remotely piloted research aircraft developed to study some of the flight characteristics of the BWB concept. It has a sophisticated flight control system under which the flight computer translates pilot commands and sensor feedbacks into deflections of hinged control surfaces across the entire trailing edge of the airplane. The stability and handling qualities of the airplane are such that a pilot would be unable to fly it without the active control system. This airplane, which was designed, built, and tested at Stanford University under the direction of Dr. Ben Tigner, provides an excellent example of the kind of unconventional configuration that many current

design optimization routines are not properly equipped to handle. As such, it provides a valuable development and testing platform for this research program. The 3-d analysis routines developed under this JIAA program during the past year played an integral role in the design, simulation, testing, and successful flight of the BWB-17.

The 3-d aerodynamic code was used to perform three essential tasks in the development of the BWB-17: α sensor and trim correction, α -limiter design, and ground effect computation. One of the key phases of testing in the BWB-17 was 'truck testing'. The airplane was mounted on a universal joint at its center of gravity, free to rotate in any direction, on top of a mast that extended out of the roof of a car. The car was then driven at flight speeds while the pilot 'flew' the plane from a chase vehicle. This testing procedure was used to tune the control system, set feedback gains, and gain confidence in many of the flight systems in a relatively low risk environment. However, there were significant differences in the airflow around the airplane on top of the car versus the airplane in free air. The 3-d code was the key element in resolving these differences so that the appropriate changes could be made to prepare the airplane for free flight.

The first main difference that had to be resolved was the sensed angle of attack. The BWB-17 is equipped with a weathervane-like instrument protruding about one foot from its nose that feeds back α and β to the computer during flight. Significant differences between the geometric angle of attack, the angle of attack measured in free air, and the angle measured on the car were computed using the aerodynamic code and this information was used to correct the flight control system (Figure 5.4). One of the most important aspects of this calibration was the design of the α -limiter. In order to reduce the risk of stall and loss of control, the BWB-17 control system includes a feedback on angle of attack and pitch rate to prevent the airplane from exceeding the predetermined upper limit on α . This limit is enforced by the flight computer, which does not allow the elevator deflection to exceed some predetermined limit at any given angle of attack. Essentially, once the α -limit is reached, the pilot no longer has the authority to pitch up. The limiter was originally designed and calibrated using the data and flow visualization (with surface tufting) from the truck testing. When this limiter was implemented on the non-linear flight simulation of the airplane in free air, however, significant problems arose, with the pilot finding that he was unable to pull out of some high speed dives. The aerodynamic design code was used to simulate both the airplane on the car and the airplane in free flight (Figure 5.5). The results showed a very large difference in the aerodynamic pitching moment on the airplane on and off the car (Figure 5.6), which resulted in roughly a 10 degree offset in elevator setting. Additionally, since the lift distribution was modified by the presence of the car, the code was used to determine what angle of attack would yield the same local maximum lift coefficient at the critical wing station. This insured that the wing would stay within the limits set by the truck stall tests. With this new data was incorporated, the

limiter worked exactly as desired. In addition to correcting the α -limiter error, this data also allowed the pilot to adjust the transmitter trim settings, which had been determined during truck testing, to the proper settings for free flight.

The final area in which this research program contributed to the success of the BWB-17 was in the computation of ground effect. Since the BWB configuration has a very high t/c center section, a simple vortex-lattice evaluation of ground effect was insufficient to determine the aerodynamic performance of the airplane during takeoff and landing. The 3-d code was used to determine the lift and moment variations with angle of attack and height in ground effect, and this data was incorporated into the non-linear simulation of the airplane. Using this new model in conjunction with the flight simulator, the pilot was able to become comfortable with the very unusual behavior of this aircraft during take-off and landing.

Throughout the development of the BWB-17, the aerodynamic code developed under this research program proved to be an excellent resource for analysis and design of this unconventional configuration. The BWB-17 recently made a very successful debut flight, and will be flown extensively in the coming year to further explore the low speed and high angle of attack performance of the BWB configuration.

2.5.4 Research Activities Proposed for 1997-1998

Some improvements remain to be made on the 3-d inviscid aerodynamic code. Now that the general algorithm has been established and validated, the code will be cleaned up to run as efficiently as possible in an optimization framework. A rational, efficient method for modifying wing geometries with a reasonable number of design variables must be developed. This method should allow adequate freedom for the optimizer to explore the design space thoroughly, but without 'micro-optimizing' every point on the wing surface.

The next step in the development of the aerodynamic design module will be the addition of a viscous model. A 2-d integral method that includes the local effects of sweep and taper on boundary layer development has been identified, and implementation of this method is currently under way. Combined with the 3-d velocity distributions from the panel code, this boundary layer code should provide a significant improvement over the current viscous model. The panel code will be modified to account for the presence of the boundary layer, increasing the accuracy of lift and induced drag computations. The existing maximum lift prediction methods will be combined with the viscous calculations to form a systematic method for evaluating maximum usable lift.

Once the aerodynamic module is completed, it will be incorporated into a full multi-disciplinary aircraft design package, such as WingMOD. This research program will continue to combine efforts with the BWB program. The collaboration provides a valuable resource to the BWB design team, while allowing

this program to take advantage of the many resources that NASA and McDonnell-Douglas devote to the BWB project. Comparisons will be made between the inviscid/viscous coupled code and full Navier-Stokes computations, as well as available experimental results from the BWB-17 flight test program. The unconventional configuration of the BWB will provide an excellent test platform for the newly developed aerodynamic design package.

2.5.5 Research Activities Planned Beyond 1997-1998

The primary goal of this research is to design better airplanes faster, and a fast accurate analysis used in conjunction with numerical optimization methods is the key to accomplishing that goal. Several other useful design tools could also come as by-products of the development and testing of this aerodynamic design package. A comparison of boundary layer properties on swept and unswept wings at similar conditions could be made to assess the applicability of using a 2-d boundary layer analysis on swept wings. This could be used to determine whether or to what extent the independence principle applies to the flow at conditions near maximum lift. Direct comparisons of velocity profiles, displacement thicknesses, momentum thicknesses and other boundary layer properties could be made between the 2-d wing, and along several cuts of a swept wing (such as normal to the sweep axis, aligned with the free-stream velocity or aligned with the inviscid streamlines). These comparisons could be used to assess the validity of using 2-d maximum lift data, and could lead to appropriate corrections to 2-d data based on properties such as inviscid pressure distributions or sweep and other geometric parameters. An attempt to build a Stratford like criteria for separation on a 3-d lifting surface could be made using data obtained from the research described above. In such a criteria, separation would be correlated with various properties of the flow such as pressure distribution, Reynolds number based on the distance the boundary layer has had to develop and the pressure gradient in the direction of local velocity. If such a criteria could be developed, it presents the best possibility for a fast but general method for preliminary assessment of maximum lift. Another key factor in modern wing design that would benefit from this improved high lift modeling is the prediction and alleviation of flutter. As the preliminary design code is refined to better assess wing high lift characteristics, a flutter analysis could be included. This would make the wing design algorithm substantially more valuable in the early stages of aircraft synthesis and optimization.

2.5.6 References

- [5.1] Wakayama, S., "Lifting Surface Design Using Multidisciplinary Optimization," Ph.D. Dissertation, Stanford University Department of Aeronautics and Astronautics, Stanford, 1994.
- [5.2] Jones, R.T., "Wing Theory," Princeton University Press, Princeton, NJ, 1990.
- [5.3] Torenbeek, E., "Synthesis of Subsonic Airplane Design," Delft University Press, Delft, Holland, 1982.

- [5.4] Furlong, G. Chester, and McHugh, James G., "A Summary and Analysis of Low Speed Longitudinal Characteristics of Swept Wings at High Reynolds Number," NACA Report 1339, 1957.
- [5.5] Letko, William, and Goodman, Alex, "Preliminary Wind Tunnel Investigation at Low Speed of Stability and Control Characteristics of Swept Back Wings," NACA TN 1046, 1946.
- [5.6] Brune, G.W., and McMasters, J.H., "Computational Aerodynamics Applied to High-Lift Systems," Applied Computational Aerodynamics, Vol. 125 of Progress in Astronautics and Aeronautics, AIAA, Washington, D.C., 1990.
- [5.7] Wakayama, S., Page, M., Liebeck, R., "Multidisciplinary Optimization on an Advanced Composite Wing," AIAA 96-4003, 6th AIAA/NASA/ISSMO Symposium on Multidisciplinary Analysis and Optimization, Sept. 1996.
- [5.8] Shevell, R.S., "Fundamentals of Flight", Prentice-Hall Inc., Englewood Cliffs, NJ, 1989.

2.5.7 Figures

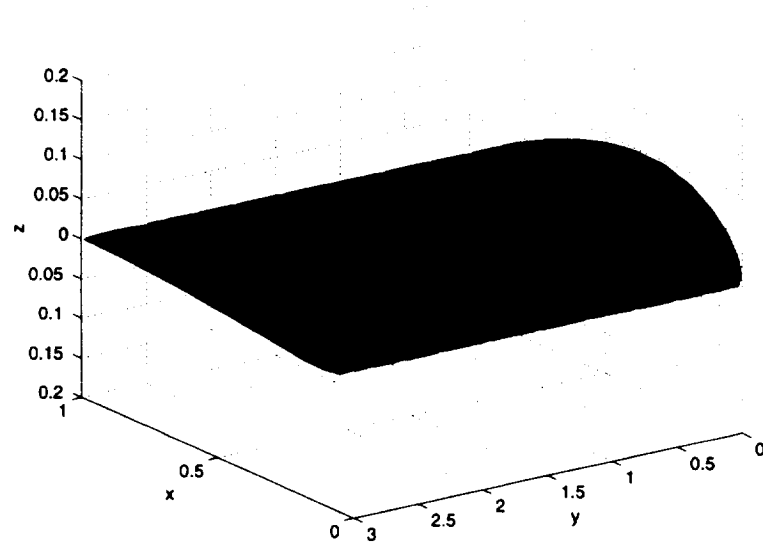


Figure 5.1 Source panel distribution for 3-d mean vortex panel code. NACA 4412, unswept, untapered wing; $y=0$ is the wing plane of symmetry

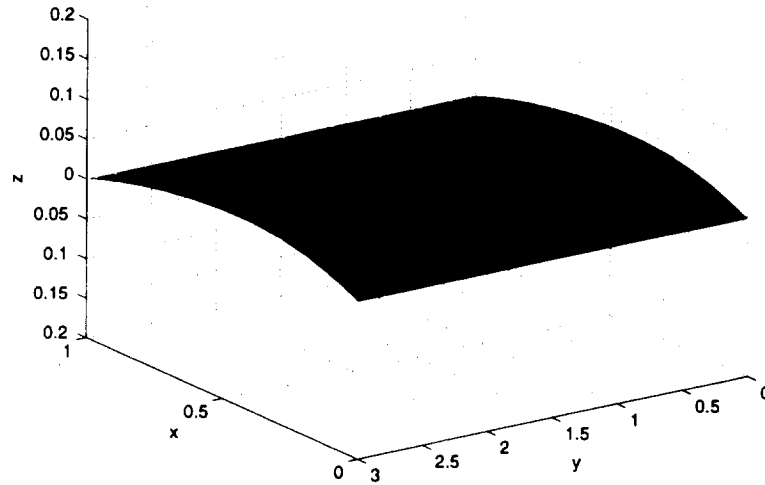


Figure 5.2 Vortex panel distribution for 3-d mean vortex panel code. NACA 4412, unswept, untapered wing

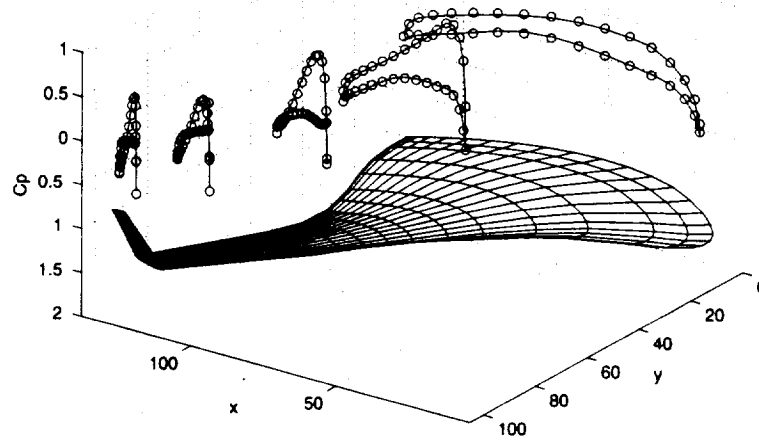


Figure 5.3 Comparison of surface pressure coefficients on BWB wing compute using A502 and the research code. Solid lines are A502 data, circles are computed from the mean surface vortex code.

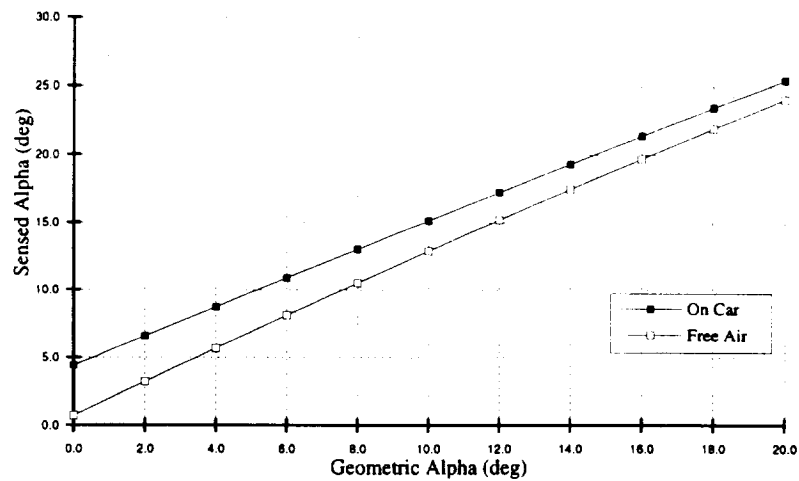


Figure 5.4 Variaton of sensed angle of attack vs. geometric angle of attack on the BWB-17 in free air and mounted on the test car.

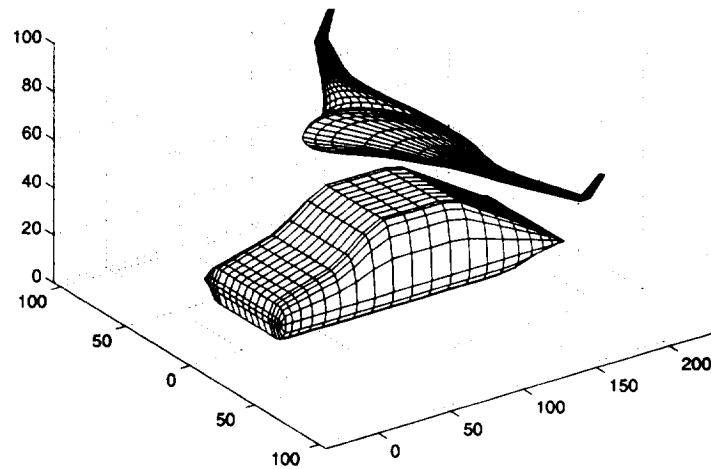


Figure 5.5 Computational grid used to analyze the aerodynamic effects of the test car on the BWB-17.

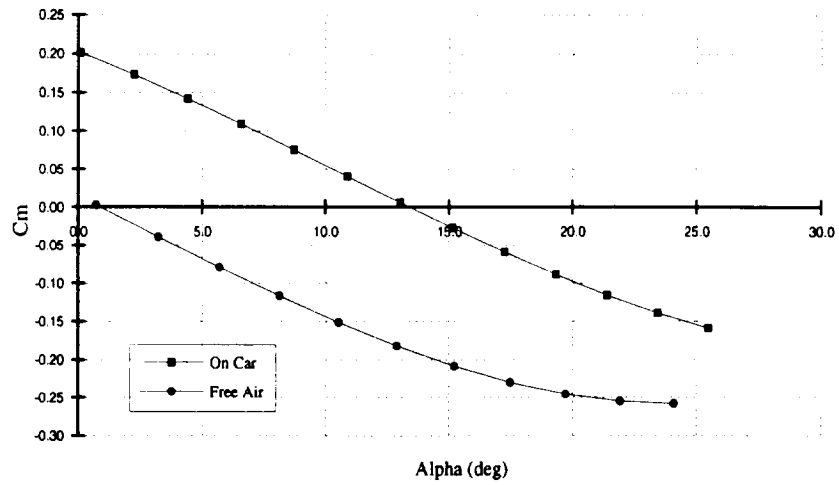


Figure 5.6 Aerodynamic pitching moment for the BWB-17 in free air and on the test car.

2.6 PROJECT 6 - ACOUSTIC WAVE SCATTERING THROUGH A TURBULENT MIXING LAYER

Research Participants: Prof. Sanjiva Lele, Graduate Student

Ames Technical Contact: Dr. N. Mansour

2.6.1 *Introduction*

In order to construct acoustic noise maps, various source localization techniques using microphones have been developed (Ref. [6.1], [6.2]). With noise source maps, the dominant noise sources can be identified which helps to reduce the noise. In this approach to noise reduction, model-scale tests are often conducted in subsonic open jet wind tunnels. The acoustic wave direction and intensity distribution is measured by the microphones located outside of the flow. Hence the noise produced by the aerodynamic flow of interest has to propagate through a turbulent mixing layer before it is received by the microphones.

A simple refraction model using a vortex sheet theory can predict the changes in the sound propagating through a shear layer fairly well at low frequencies. The combination of vortex sheet and ray tube theory provides a good prediction of the acoustic wave amplitude (Ref. [6.3], and Figure 1). In this regime, we have also studied the refraction phenomena more accurately by accounting for the finite thickness of the shear layer via the use of geometric acoustic approximation. In this refined model, the ray tracing equations (the eikonal equations) give the refracted ray path and a higher approximation (the transport equation) gives the acoustic amplitude. The predictions are shown in Figure 2 and 3, which also compare them to the experimental data and the simpler vortex sheet model. While in the present regime the refined theory shows no significant deviation from the vortex sheet model, it is anticipated that such coincidence will break down in other regimes. For example at high frequencies (10 kHz or higher), where the wave length is comparable or smaller than the thickness of the shear layer, the theory does not show satisfactory agreement with the experimental data. A thorough investigation of the scattering due to shear layer turbulence and development of an approximate theory to predict the scattering, are the main goals of this proposed research.

2.6.2 *Research Objective*

To quantitatively predict the scattering phenomena, it is necessary to understand its mechanism and the effects of the relevant non-dimensional flow parameters and geometric conditions. In past, several theories (Ref. [6.4], [6.5], [6.6]) have been developed to estimate the acoustic scattering from turbulence. These studies made the assumption that the turbulence is isotropic and the scattering is mainly due to turbulent motions with larger spatial scales compared with the acoustic wave length. Experiments (Ref. [6.7], [6.8]) have shown that the turbulent-nonturbulent interface plays a significant role in the scattering and the importance of turbulent motions at the Taylor microscale has been noted. In view of the experimental findings, our focus is on the case where

the energy containing peak scale of the turbulence is comparable to the acoustic wave length. In this regime, neither high nor low frequency limit cannot be employed, and a systematic study using numerical simulations is needed.

In this research, we propose to construct a reliable method to represent the acoustic scattering numerically. First, this method will be used to investigate the effects of scattering from large, intermediate and small scale spectral ranges of turbulence. With this information, approximate methods which allow good quantitative predictions of the scattering will be developed.

2.6.3 Research Program

The first step is to develop a LES code which can calculate both the turbulent flow in a shear layer and acoustic fields of interest simultaneously, so that it can be used to study scattering due to turbulence accurately. The simulation will duplicate the conditions of the Schlenger and Amiet experiments (Ref. [6.3]), and its results will be compared to the experiments in detail to establish that an accurate simulation of the scattering from turbulence has been achieved. Once this has been accomplished, other approaches to predict scattering will be examined. One approach which we plan to evaluate is based on formulating the scattering problem in terms of a radiation problem (with a third order wave equation - Lilley's equation) and stochastic source terms representing the interaction of the shear layer turbulence with the incident wave. A similar approach was developed by Lighthill (Ref. [6.4]) and has been quite successful in describing the scattering for simpler flows. The LES data will be used to provide the information on the stochastic scattering sources. Other approaches based on geometric acoustics will also be evaluated.

2.6.4 Research activity proposed for 1997-1998

First, a LES code suitable for the scattering problem will be developed. An existing Direct Numerical Simulation (DNS) code developed by J. Freund (1997) will be adapted for this purpose. This code has been used to directly simulate the sound radiation due to a supersonic jet. Its ability to provide good resolution for sound waves is well demonstrated. The code provides robust boundary conditions for turbulent inflow. Subgrid models will be incorporated into this code. Our plan is to use the dynamic models of subgrid scales developed at CTR (Ref. [6.9]). The predictions of the LES code will be compared with the experimental data.

2.6.5 Research activity planned beyond 1997-1998

Once we have established the reliability of the LES code, parametric studies of the experimental conditions will be undertaken. Using Fourier transform, we plan to extract the statistical information about the scattering sources and obtain a scattering prediction by solving Lilley's equation. The effects of small and large scale turbulence and the interface motion will be studied. Through this procedure, the principal factor responsible for scattering will be identified. To extend scattering analysis to more realistic noise sources, we propose to

solve the scattering problem for dipole and quadrupole sources. In the applications of acoustic arrays, such features of noise sources need to be distinguished from distributed noise sources. We believe this research will help in further advancement of noise measurement technologies.

2.6.6 *References*

- [6.1] Mosher, M., "Phased Arrays for Aeroacoustic Testing: Theoretical Development," AIAA 96-1713, 1996.
- [6.2] Sen, R., "Interpretation of Acoustic Source Maps made with an Elliptic-mirror Directional Microphone System," AIAA 96-1712, 1996.
- [6.3] Schlinker, R. H., and Amiet, R. K., "Shear Layer Refraction and Scattering of Sound," AIAA 80-0973, 1980.
- [6.4] Lighthill, M. J., "On the energy scattering from the interaction of turbulence with sound or shock waves," Proc. Camb. Phil. Soc., 1953, 49, 531-551.
- [6.5] Tatarski, V. I., Wave propagation in a turbulent medium, translated by Silverman, R. A., 1960, McGraw-Hill, New York.
- [6.6] Keller, J. B., "Wave propagation in random media," Proc. Sym. Appl. Math. 1964, 16, 227-246.
- [6.7] Candel, S. M., Jullanad, M., and Jullienne, A., "Shielding and scattering by a jet flow," AIAA 76-545, 1976.
- [6.8] Ho, C. M., and Kovasznay, L. S. G., "Propagation of a coherent acoustic wave through a turbulent shear layer," J. Acoust. Soc. Am., 1978, 60, No. 1, 40-45.
- [6.9] Ghosal, S., Lund, T. S., Moin, P., and Akselvoll, K., "A dynamic location model for large-eddy simulation of turbulent flows," J. Fluid Mech., 1995, 286, 229-255.

2.6.7 Figures

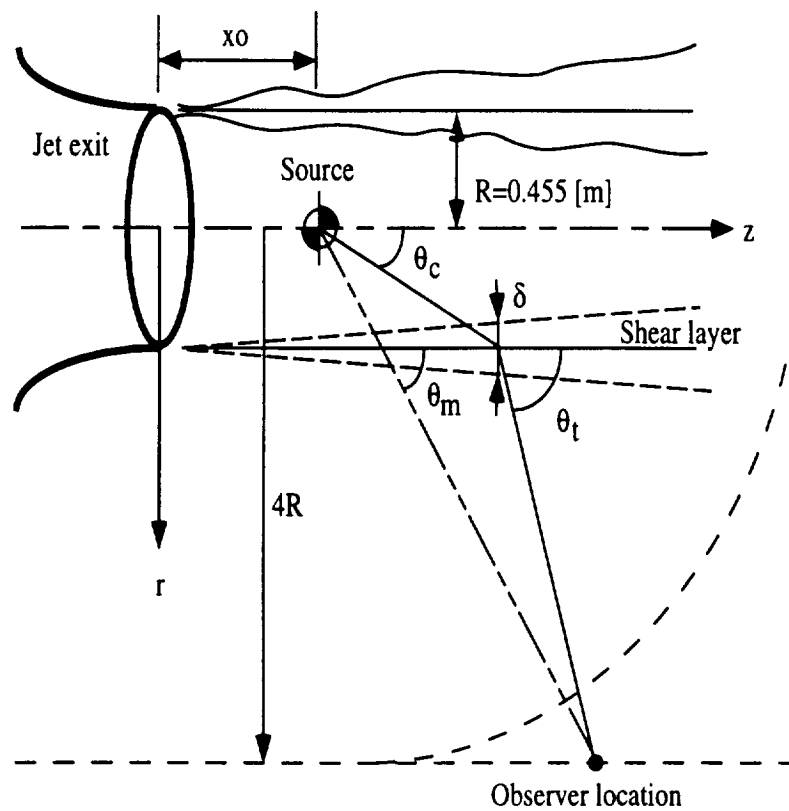


Figure 6.1 Experimental geometry by Schlinker and Amiet 1980 (Ref. [3])

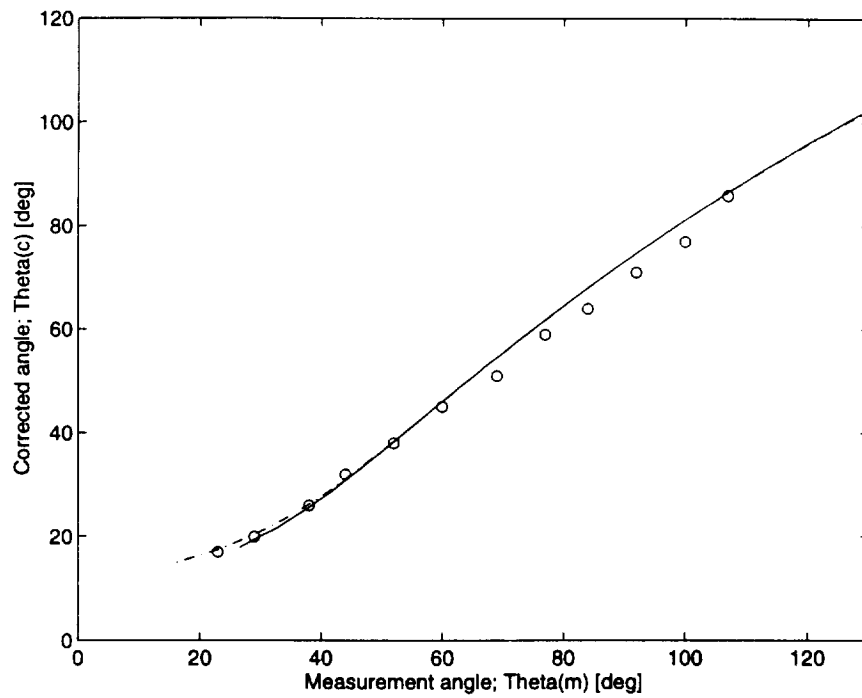


Figure 6.2 Comparison of predicted refraction angle correction through a shear layer with experiment at $M=0.4$, $f=10\text{kHz}$, $d=0.45R$ (0.2m)

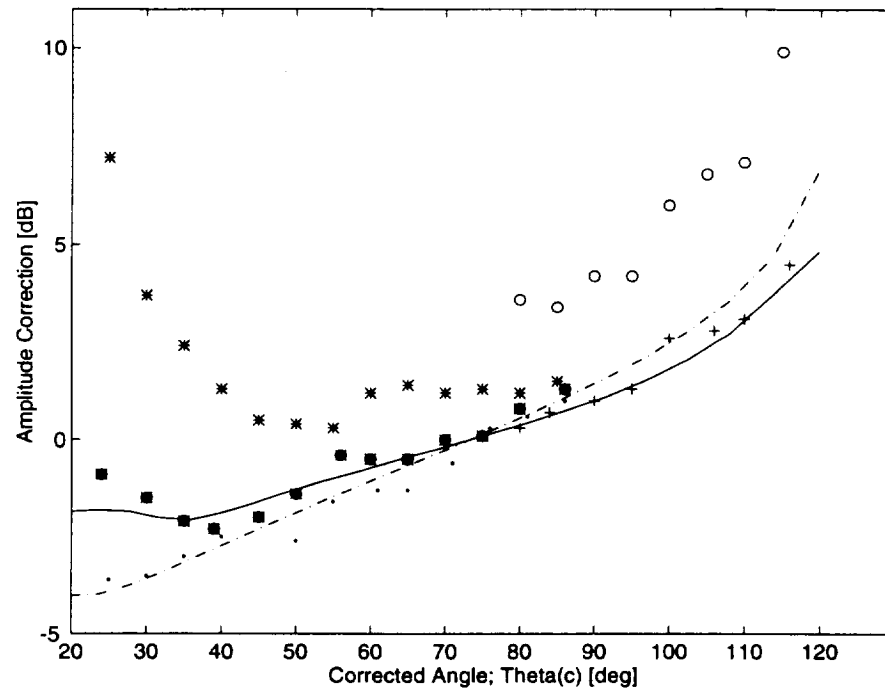


Figure 6.3 Comparison of predicted refraction amplitude correction through a shear layer with the experiments at $M=0.3$ (Sclinker and Aimiet 1980)

3. PERSONNEL

The JIAA program described above will involve faculty, staff and graduate students as follows:

Faculty:

Prof. Brian Cantwell (PI), 15% academic year, 25% summer

Prof. Steve Rock, 5% academic year, 5% summer

Prof. Sanjiva Lele, 5% academic year, 5% summer

Prof. Ilan Kroo, 5% academic year, 5% summer

Staff:

Research Assistants: 6 Ph.D. students 50% AY/100% summer

Sci-Eng. Associate 10% calendar year

Admin. Associate 20% calendar year

Throughout the conduct of this research and training activity close coordination will take place between the research personnel at Stanford and the research personnel and technical management staff at Ames.

4. FUNDING

The funding requested for the one-year period, Ref. October 1, 1997 to September 30, 1998 is given in the attached Estimated Cost Breakdown. As the University's contribution to the administration of the Institute, indirect costs on Professor Cantwell's administrative salary charges and administrative and secretarial support are waived.

JIAA 97-98 Admin Bud

ESTIMATED COST BREAKDOWN - JIAA 97-98 ADMINISTRATIVE BUDGET

GRANT NUMBER NCC 2-55

PROPOSAL NUMBER - AERO 97-52

DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1997 TO SEPTEMBER 30, 1998

	<u>ACADEMIC SUMMER CALENDAR</u>		<u>COST</u>
A. SENIOR PERSONNEL			
B. Cantwell, Prof.	10.00%	20.00%	14,480
B. OTHER STAFF			
C. Edwards, Admin. assoc., coordinates JIAA project supervision	15.00%		5,152
TOTAL SALARIES and WAGES (A+B)			<u>19,632</u>
C. FRINGE BENEFITS (applied to TOTAL SALARIES AND WAGES)			
26.2% through 9/30/98			5,144
TOTAL SALARIES, WAGES and FRINGE BENEFITS (A+B+C)			<u>24,776</u>
D. SUB-TOTAL DIRECT COSTS (A+B+C)			24,776
E. MODIFIED TOTAL DIRECT COSTS (D)			24,776
F. UNIVERSITY INDIRECT COSTS (Waived)			
0.00% through 9/30/98			
G. ANNUAL AMOUNT REQUESTED (D+F)			<u>24,776</u>
TOTAL PROJECT COST			<u><u>24,776</u></u>
TOTAL ESTIMATED JIAA COST FOR 97-98			
Admin Budget			24,776
On-campus Budget			363,293
ESTIMATED TOTAL JIAA 97-98 BUDGET			<u><u>388,069</u></u>

ESTIMATED COST BREAKDOWN - JIAA 97-98 ON-CAMPUS BUDGET

GRANT NUMBER NCC 2-55

PROPOSAL NUMBER - AERO 97-52

DURATION - 12 MONTHS BEGINNING OCTOBER 1, 1997 TO SEPTEMBER 30, 1998

ACADEMIC SUMMER CALENDAR COST

A. SENIOR PERSONNEL			
B. Cantwell, Prof.	5.00%	10.00%	7,240
S. Rock, Assoc. Prof.	5.00%	10.00%	6,867
I. Kroo, Assoc. Prof.	5.00%	10.00%	6,807
S. Lele, Assist. Prof.	5.00%	10.00%	6,094
B. OTHER STAFF			
V. Matte, Sci.-Eng. Assoc.		10.00%	6,536
TOTAL STAFF SALARIES and WAGES (A+B)			<u>33,544</u>
C. FRINGE BENEFITS (applied to A+B)			
26.2% through 9/30/98			8,789
TOTAL STAFF SALARIES, WAGES and FRINGE BENEFITS (A+B+C)			<u>42,333</u>
D. STUDENTS			
Student research assistant	50.00%	50.00%	18,360
Student research assistant	50.00%	50.00%	18,360
Student research assistant	50.00%	50.00%	18,360
Student research assistant	50.00%	50.00%	18,360
Student research assistant	50.00%	50.00%	18,360
Student research assistant	50.00%	50.00%	18,360
TOTAL STUDENT SALARIES			<u>110,160</u>
E. OTHER COSTS			
Communications			1,500
Report costs, postage, distribution of special reports			4,000
Travel - 4 trips to an east coast conference			4,000
travel and living expenses for a visiting scientist for one month			8,000
TOTAL OTHER COSTS			<u>17,500</u>
F. CAPITAL EQUIPMENT			
Project 1 - control system fabrication-2k			2,000
Project 2 - Data display system-5k			5,000
Project 3 - h.w. anemometers-17k, press. transducers(2)-5k, press. calibrator-5k			27,000
Project 4 - Image capture board-5K, digital camera-10K, UVlamps(2)-2k			17,000
TOTAL CAPITAL EQUIPMENT			<u>51,000</u>
G. GRADUATE STUDENT TUITION			
three students, 50% units @ 12128 for 4 quarters			36,384
three TGR students @ 3460 for 4 quarters			10,380
TOTAL TUITION			<u>46,764</u>
H. TOTAL COSTS NOT SUBJECT TO INDIRECT COST (F+G)			97,764
I. TOTAL COSTS SUBJECT TO INDIRECT COST (C+D+E)			169,993
J. UNIVERSITY INDIRECT COST (56.2*1)			
56.2% THROUGH 9/30/98			95,536
TOTAL PROJECT COST (H+I+J)			<u><u>363,293</u></u>